NASA Contractor Report 172406

P-144

Study on Utilization of Advanced Composites in Fuselage Structures of Large Transports

Final Report

R. W. Johnson, L. W. Thomson, R. D. Wilson

Boeing Commercial Airplane Company
P.O. Box 3707
Seattle, Washingotn 98124

IN-24
117205
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Contract NAS1-17417

February 1985

(NASA-CR-172406) STUDY ON UTILIZATION OF ADVANCED COMPOSITES IN FUSELAGE STRUCTURES OF LARGE TRANSPORTS Final Report (Boeing Commercial Airplane Co.) 144 p CSCL 11D

N88-15014

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G3/24 0117205

National Aeronautics and Space Administration

Langley Research Center
Hampton, Virginia 23665

FOREWORD

This final technical report was prepared by the Boeing Commercial Airplane Company, Renton, Washington, under NASA Contract NAS1-17417. This report covers work performed between May 1983 and May 1984. The program was sponsored by the National Aeronautics and Space Administration, Langley Research Center (NASA-LRC), and the Air Force Wright Aeronautical Laboratories. Herman L. Bohon was the NASA-LRC ACEE project manager and Jon S. Pyle was the NASA-LRC technical manager. J. L. Mullineaux (AFWAL) was the Air Force technical manager.

The following Boeing personnel were principal contributors to the program:

ACDP Program Director
S. T. Harvey

Program Manager
R. D. Wilson

Technology Manager
R. W. Johnson

Structural Analysis
B. R. Frakes
R. Khanna
L. W. Thomson

Design
M. J. Thould

Weight and Balance Analysis
J. T. Parsons
T. S. Whitehurst

Materials
W. C. Potter

Systems
D. G. Moss

Manufacturing Technology
H. R. Fenbert
P. K. Hanks
G. A. Jump

Business Management
R. E. Sever

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SYMBOLS AND ABBREVIATIONS

a_o	characteristic dimension
ACEE	aircraft energy efficiency
B	panel width
BL	buttock line
C	core thickness
C_L	centerline
D	diameter
DUL	design ultimate load
E	modulus of elasticity
E_{11}	lamina modulus of elasticity in fiber direction
E_{22}	lamina modulus of elasticity in transverse direction
EI	bending stiffness
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulation
FT	feet
g	acceleration constant (32 ft/sec2)
G	shear modulus
gal	gallon
G_{12}	lamina shear modulus
hr	hour
IML	inside mold line
IR&D	independent research and development
K	empirical correction factor for temperature, moisture, pressure, and curvature
KIP	10^3 lb
KSI	10^3 lb/in^2

SYMBOLS AND ABBREVIATIONS (Continued)

L	column length
LB	pounds
LEMP	lightning electromagnetic hardening
LRU	line replacement unit
M	ultimate bending moment
msi	$1x10^6$ pounds per square inch
N	end load
nmi	nautical miles
Nx	longitudinal end load
Ny	circumferential end load
Nxy	shear load
n	number of plies
NC	numerically controlled
OML	outside mold line
P	load
P_B	bearing load
psi	pounds per square inch
q	shear flow
R	radius
S	percent stiffening. In laminate code, designates symmetry.
SEC	body section
STA	station identification along longitudinal direction of fuselage
sym	symmetric
t	laminate thickness
$\bar{t}$	smeared thickness

SYMBOLS AND ABBREVIATIONS (Continued)

TTU	through-transmission ultrasonic
V	ultimate shear load
W	fastener spacing
WL	waterline station identification along vertical direction of fuselage cross section
x	longitudinal direction along fuselage
y	circumferential direction on fuselage
α	bearing load angle
γ	shear strain
ε_C	compression strain allowable
ε_T	tension strain allowable
θ	laminate angle
ν_{12}	lamina Poisson's ratio
ϱ	density
σ	stress
σ_B	bearing stress
σ_x	longitudinal stress

1.0 INTRODUCTION AND SUMMARY

Several recent NASA- and DOD-sponsored programs have shown that using advanced composites in aircraft structures, especially primary structures, can result in significant weight reductions with ensuing fuel economy improvements. The potential benefits of applying composites to fuselage structure are as significant as those of applying composites to wing structure, since the wing and fuselage account for approximately equal fractions of the aircraft structural weight (fig. 1.0-1). Additional benefits can be realized by applying composites to fuselage structure, because weight reductions at the airplane centerline are more effective in increasing payload due to the offsetting deadweight relief effects (fig. 1.0-2).

In addition to weight reduction, applying composites to fuselage structure will reduce fabrication costs. Relative to the other major airframe components, metal fuselage components are the most expensive per pound of structure (fig. 1.0-3). These high costs are due to the high part count (fig. 1.0-4) and resulting assembly expense. In a composite fuselage shell, the part count can be reduced by approximately 20% of an aluminum shell part count by the use of cocured composite components such as skins and stringers and/or honeycomb bonded assemblies.

Operational and maintenance costs will be lower for composite airframes due to a reduction in part count, improvements in fatigue performance, and corrosion resistance. The fuselage typically has the highest percentage of fatigue problems compared to other components. Fatigue problems are one of the major contributors to repair and maintenance costs. Application of fatigue-resistant composite materials to the fuselage has the potential to reduce these costs substantially. In addition, use of corrosion-resistant composite structures will reduce commercial airline and military maintenance costs in the high-corrosion areas of the fuselage.

In the current study on utilization of advanced composites in fuselage structures of large transports, the following tasks were performed:

- Selected and developed six composite fuselage design concepts
- Evaluated design concepts in terms of:
    - Structural performance
    - Weight
    - Manufacturing development and costs
- Calculated weight reduction due to composites application to the fuselage of a commercial transport
- Calculated weight reduction due to composites and aluminum-lithium alloy application to the fuselage of a military transport
- Determined benefits to a fleet of military transports
- Identified and evaluated significant technology issues pertinent to composite fuselage structures
- Developed program plans for resolving technology issues
- Selected Boeing's preferred option for demonstrating technology readiness

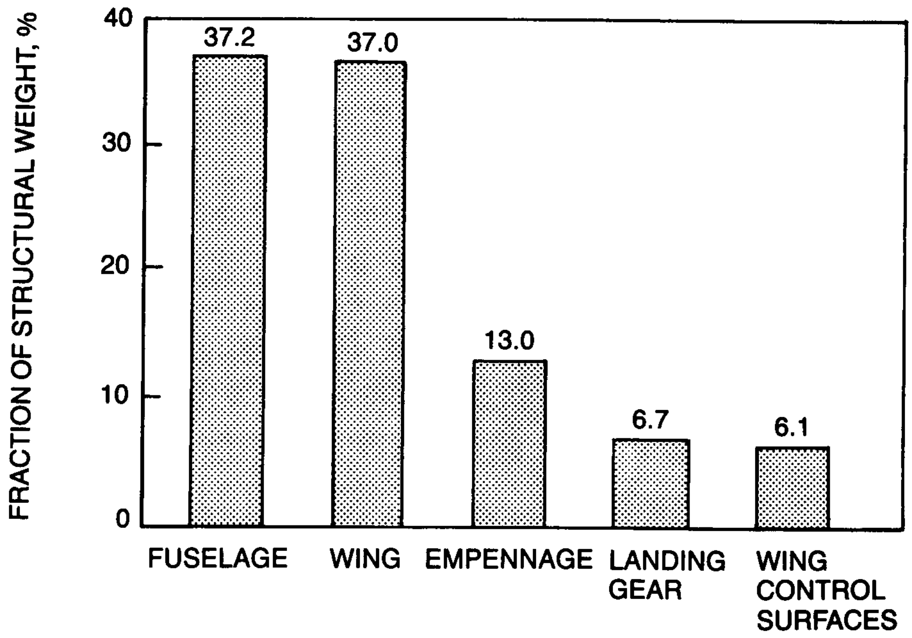

Figure 1.0-1. Typical Commercial Transport Component Weight Distribution

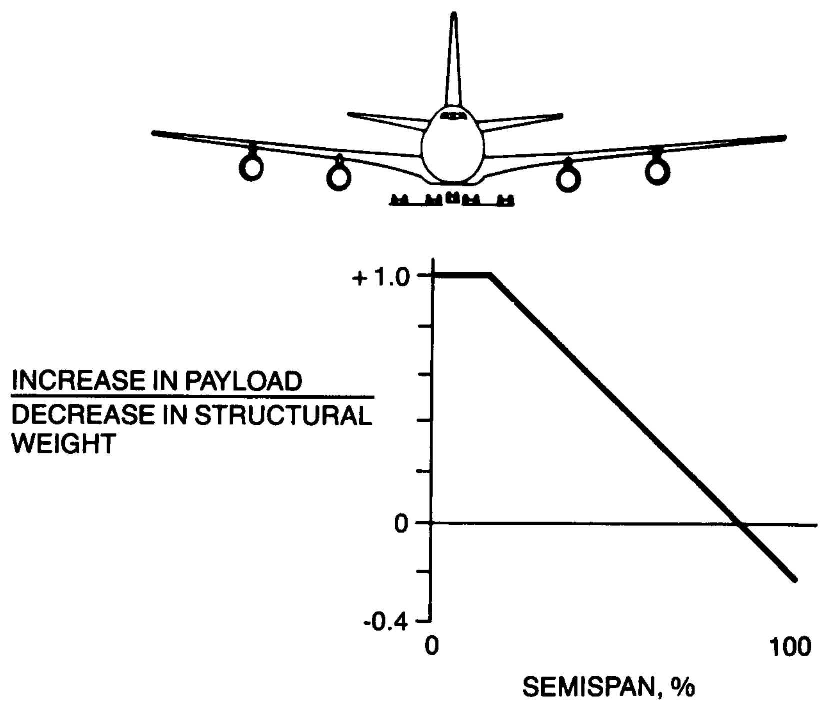

Figure 1.0-2. Centerline Payload — Weight Effect

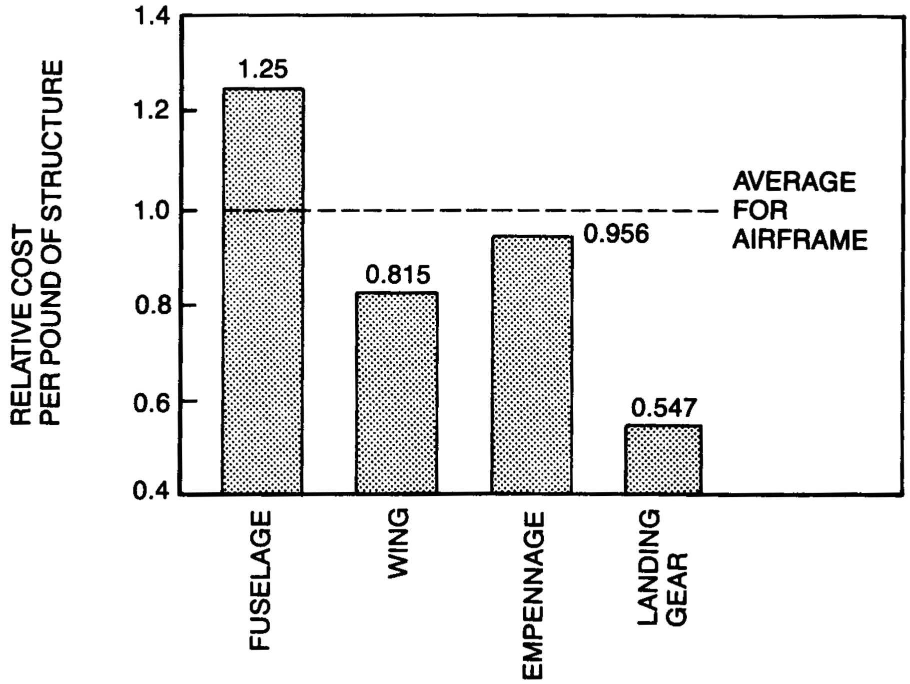

Figure 1.0-3. Typical Commercial Transport Component Cost Comparison

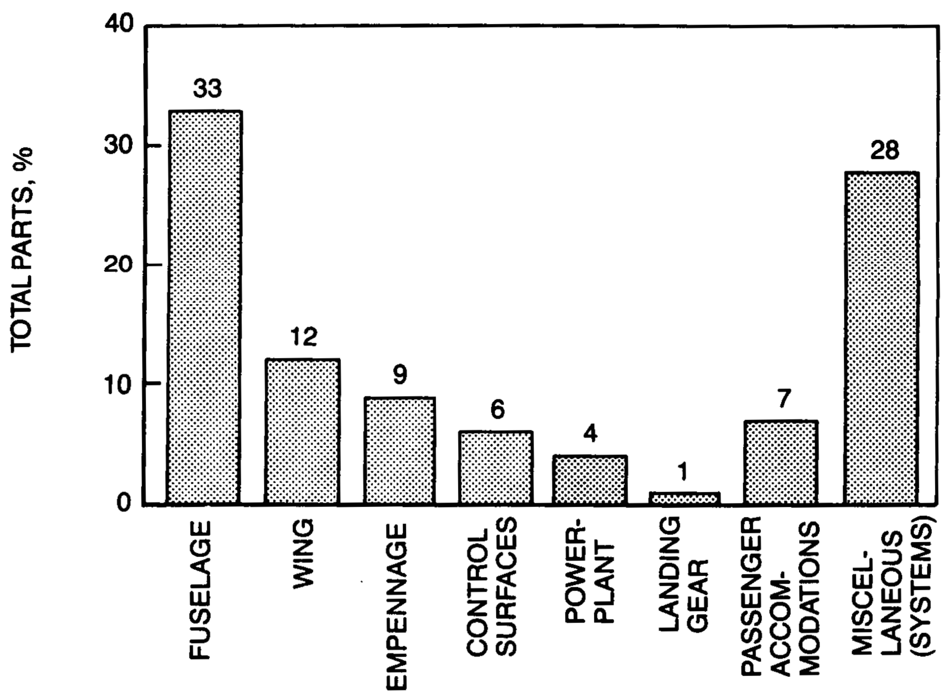

Figure 1.0-4. Typical Commercial Transport Part Count Distribution

The study of potential benefits of applying composite materials to the fuselage was initiated by developing and evaluating six diverse fuselage shell concepts. The concepts ranged from a stiffened skin configuration to an unstiffened honeycomb shell. The study demonstrated that the extensive use of composite materials in an aftbody fuselage section can reduce the shell weight by as much as 30%. Weight reduction studies were performed on all the commercial fuselage structure considered candidate for composite materials application. The weight reduction for this structure is approximately 21%. The weight reduction associated with applying composites to candidate structure of the fuselage of a military medium range tactical transport is estimated to be approximately 19%.

The following areas were identified as technology issues that need to be resolved in order for composite materials to be used in fuselage primary structures.

- Materials
  - Flammability and fire protection
  - Design strain levels
  - Impact damage

- Structures
  - Pressure damage containment
  - Stability and postbuckling
  - Joints, splices, and attachments
  - Cutouts
  - Impact dynamics
  - Repair

- Systems
  - Lightning protection
  - Electromagnetic effects
  - Acoustic transmission

- Manufacturing
  - Fabrication
  - Assembly
  - Quality control

Under the NASA Aircraft Energy Efficiency (ACEE) program, significant technology readiness development has been completed for composite wing primary structures. Studies performed in the current program have identified how a similar technology readiness can be achieved for composite fuselage structures by 1990. Several plans or options for achieving this degree of readiness have been developed. The execution of a selected option will provide the data base necessary to resolve the significant issues pertinent to composite fuselage structural design, fabrication, and performance.

Five program options that address the primary technology issues and provide the data base for demonstrating technology readiness have been developed. Option 1 addresses all the technology issues, except that a static and durability test of a full-scale fuselage section is omitted. Option 2 includes a static and durability test of a full-scale fuselage section, but omits large panel verification tests. Option 3 includes the large panel verification tests and a full-scale aftbody section static and durability test. Option 4 includes large panel verification tests and a full-scale fuselage center section test. Option 5 includes a flight test program of a 20-foot-long barrel section. Boeing has selected Option 3 as the preferred technology readiness plan. The program elements and the proposed schedule are shown in Figure 1.0-5.

The Boeing Company has estimated that the selected option will require an expenditure of approximately 1000 labor-years to achieve technology readiness by 1990. The estimate reflects total resource requirements regardless of funding sources, and assumes the availability of relevant data that might be available from other programs either now completed or planned concurrently with the recommended fuselage program. The estimate is a rough-order-of-magnitude (ROM) and was prepared for planning purposes only and does not represent a Boeing Company commitment.

This Advanced Composites Fuselage Study Program is an essential step in establishing the development necessary to commit advanced composite materials for commercial production of primary fuselage airplane structure by the mid-1990s.

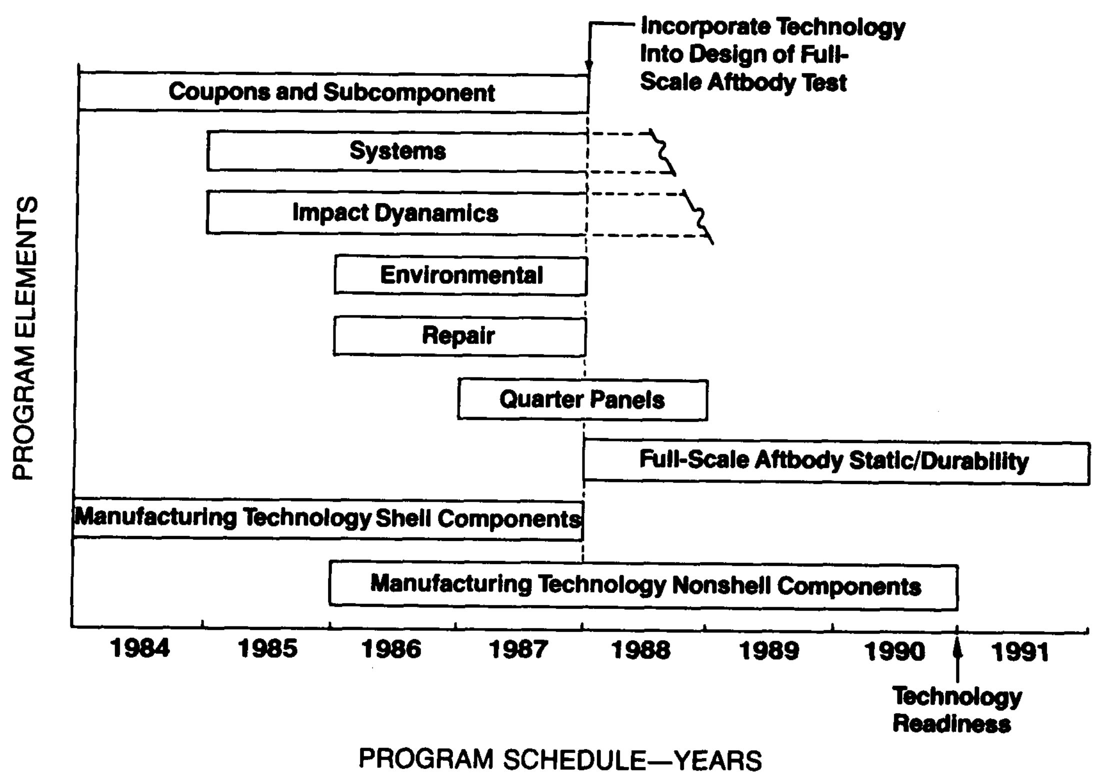

Figure 1.0-5. Boeing Proposed Fuselage Development Program

2.0 PRELIMINARY DESIGN

2.1 DESIGN CRITERIA AND GUIDELINES

The objective of the design effort has been to develop the basic configurations of six candidate composite fuselage concepts. The level of design definition included sufficient detail to evaluate the relative merits of the concepts in terms of structural performance, weight, and producibility and inspectability.

The design development was performed using the lamina properties of a graphite-epoxy tape with 35% resin content by weight, as shown below:

$E_{11} = 18.0 \times 10^6$ psi (modulus in fiber direction)

$E_{22} = 1.4 \times 10^6$ psi (modulus transverse to fiber direction)

$G_{12} = 0.98 \times 10^6$ psi (shear modulus)

$\nu_{12} = 0.34$ (Poisson's ratio)

Ply thickness = 0.0074 in

The design criteria for the composite fuselage trade study are listed below.

1. Basic material ultimate design strains:
   a. Tension $\varepsilon_T = 0.006$ in/in
   b. Compression $\varepsilon_C = 0.005$ in/in
   c. Shear $\gamma = 0.010$ in/in

2. Laminate skin elements shall be buckling resistant to 30% design ultimate load (DUL) in stringer stiffened designs. Honeycomb sandwich skin configurations shall be buckling resistant to 100% DUL.

3. The fuselage must withstand design ultimate flight loads in combination with appropriate pressure design load cases:

   - Normal operating pressure: 8.6 psi
   - Maximum pressure relief: 9.1 psi
   - Ultimate pressure with flight loads: 1.5 x 9.1 = 13.65 psi
   - Ultimate pressure only: 2.0 x 9.1 = 18.2 psi
   - Maximum damage tolerance pressure: 9.6 psi

4. The fuselage skin panels shall be damage tolerant to a 12-inch cut in any direction.

The ultimate material design strain values are based on the results of Boeing IR&D test programs. These design strain values have been validated by the NASA-funded LCPAS studies conducted by Boeing (ref. 2.1-1). These design strain values include reductions for temperature and moisture. The 30% DUL buckling criteria has been selected to provide buckle-resistant fuselage panels during normal 1-g cruise conditions. This minimizes fatigue cycling of the buckled structure and provides minimum aerodynamic drag. Other than the ultimate strain criteria and panel stability requirements, there are no special

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stiffness requirements for the fuselage compared to flutter stiffness requirements for the wing, for example. The composite fuselage has been designed to these conditions using balanced, symmetric cross-ply laminates with moduli in the range of 6 to 12 msi.

Boeing has traditionally used the 12-inch damage criterion in aluminum structures to demonstrate damage containment. This criterion allows damage to occur at any location in the skin, and to completely sever a frame or stringer. The damage is allowed to progress across the skin bay, but must be arrested at the next frame or stringer.

2.2 DESIGN EMPHASIS

The primary emphasis of the design effort has focused on the shell structure, which includes the skin, stringers, and frames. As shown in Figure 2.2-1, the shell typically accounts for 43% of the total fuselage weight of metal aircraft. In addition to the basic shell structure, attention has been given to the design of details, such as circumferential and longitudinal splices, joints and attachments, and window structure.

The design study was performed on a fuselage aftbody section. The critical loads in this section are developed primarily from down tail bending loads causing the crown to be loaded in tension and the keel in compression. The side regions are primarily sized by shear loading. The relative magnitude of load in each of the quadrants, as well as the type of loading (fig. 2.2-2), dictates the most efficient structural configuration for that particular panel.

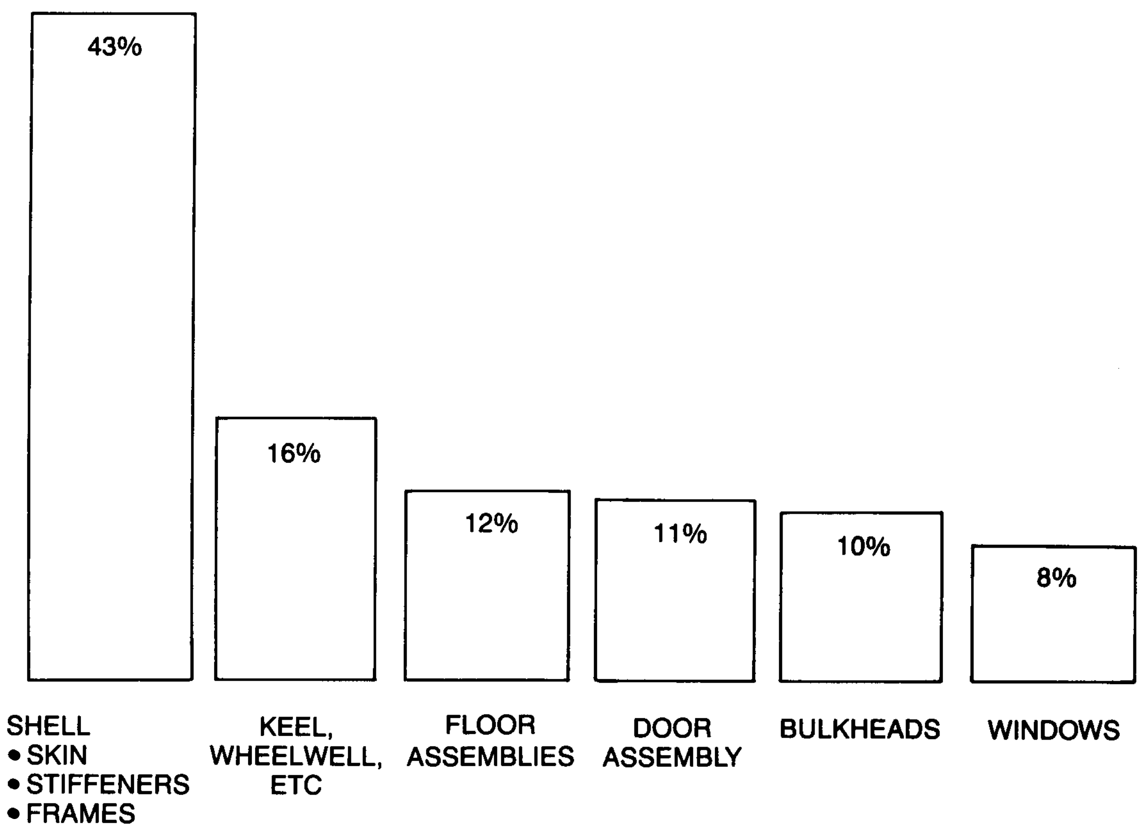
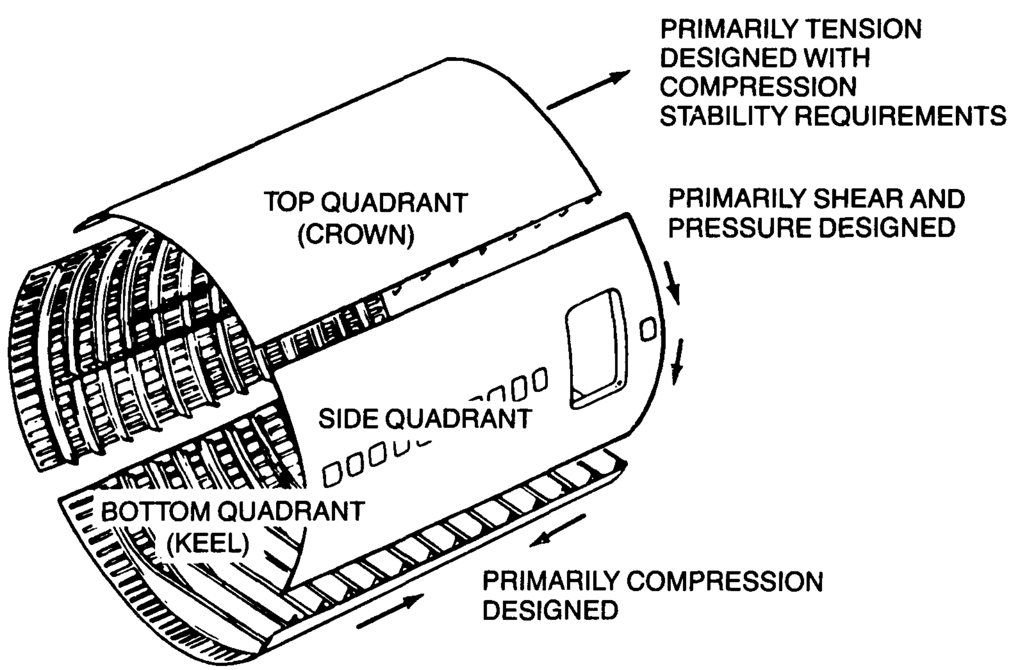


Figure 2.2-1. Typical Weight Distribution of a Commercial Transport Fuselage



FUSELAGE	QUADRANT	RELATIVE PANEL LOAD MAGNITUDE		
		TENSION	COMPRESSION	SHEAR
	TOP	HIGH	LOW	NOMINAL
	SIDE	NOMINAL	NOMINAL	HIGH
	BOTTOM	LOW	HIGH	NOMINAL

Figure 2.2-2. Typical Fuselage Construction and Major Design Parameters

2.3 DESIGN PROCEDURE

The design configurations have been sized to meet the requirements of load, strain, stability, and damage tolerance.

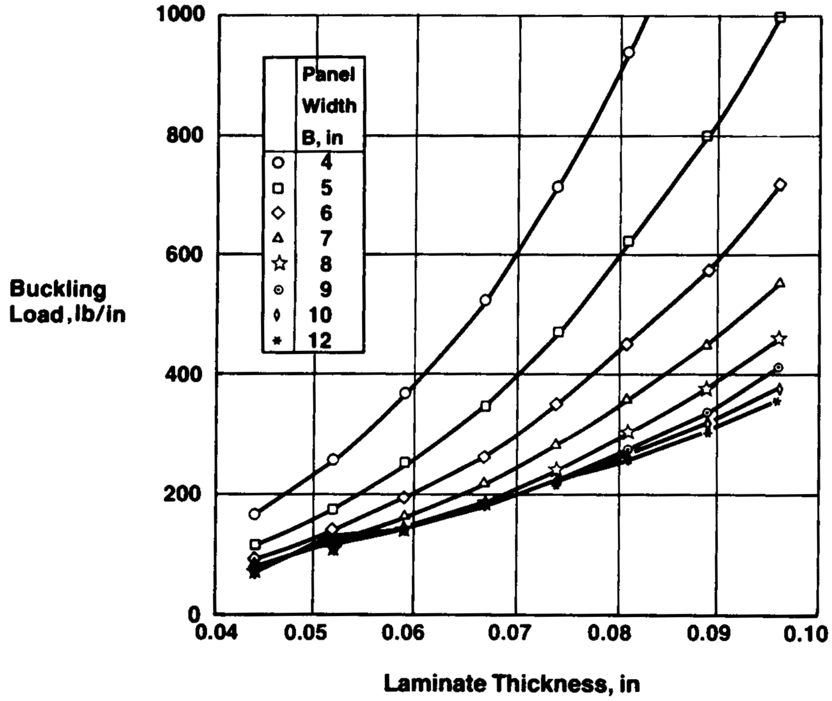
Stability requirements include local stringer buckling, skin buckling, column strength, and general fuselage cylinder stability. The stringer elements are designed to remain buckling stable to design ultimate load (DUL). The laminate skin panels are designed to remain stable until 30% DUL, and honeycomb skins are designed to be stable until 100% DUL. The load levels at which the skins buckle have been calculated based on an analysis procedure initially developed by Davenport (refs. 2.3-1, 2.3-2). This analysis has been expanded to address the orthotropic characteristics of composite laminates, and demonstrates good agreement with published analysis methods (refs. 2.3-3 through 2.3-5). Example laminate and honeycomb design curves are shown in Figures 2.3-1 and 2.3-2.

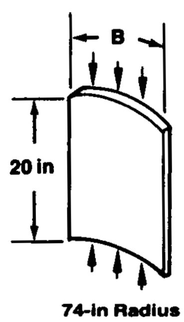
The column stability of stringer-skin elements loaded between adjoining frames has been checked using the conventional Euler column relationship. An effective width of unbuckled skin is included in the bending stiffness of the element. Since honeycomb skins are not allowed to buckle until 100% DUL, all of the honeycomb skin is considered effective and is included in the column stiffness.

The general shell stability of the fuselage was evaluated by modeling the shell as a cylinder with constant circumferential properties. The stability of the cylinder is dependent on the stiffness of the unbuckled skin, stringers, and frames, and was calculated by using the procedures described in References 2.3-6 and 2.3-7. After the skin-stringer geometries were sized to meet extensional stiffness and panel stability requirements, this analysis was used to determine stiffness and gage requirements for body frames.

Fuselage structures must be able to withstand an inflight damage located anywhere in the shell. The damage may cut through a frame or stiffener, but must be contained within the adjoining skin bays. To account for this, tear strap requirements have been developed based on a flat panel, finite-element analysis. The analysis assumes that a 12-inch damage through a tear strap will propagate and be arrested at the adjoining tear straps. The critical load for the panel is based on a critical fiber strain of 0.015 in/in at a characteristic dimension 0.10 inch beyond the crack tip. The analysis assumes that the critical fiber strain and characteristic dimension are independent of laminate orientation. This analysis procedure was initiated in a Boeing development program that modeled wing and fuselage panels with stringer elements as tear straps. The analysis procedure correlated well with Boeing IR&D testing of flat stringer stiffened panels.

In the current study program, an analysis model with tear straps at 10-inch spacing, shown in Figure 2.3-3, was developed. The analysis model contained a 16-inch cut. This damage simulates an initial 12-inch cut that has propagated and arrested at the edge of the adjacent tear straps (see sec. 2.1). The strain distribution in the crack tip region is calculated on a fine mesh grid made up with 0.04-inch by 0.04-inch elements. Several finite-element analyses were performed for different skin panel laminate orientations and percent tear strap stiffening. A similar analysis for tear straps at 20-inch spacing was performed. The design curves that were developed from these analyses, shown in Figure 2.3-4, are presented in parametric form in terms of modulus, loading, and skin thickness. A correction factor (K) is included to account for the effects of temperature, moisture, pressure, and curvature. A correction factor (K) of 0.5 was used in this study program to determine tear strap requirements. The results of Boeing IR&D allowable testing programs indicate that environmental considerations of temperature and moisture may reduce dry, room temperature strengths by 20%. Factors for out-of-plane bending and peeling effects due to curvature and pressure are not established and need to be evaluated (see sec. 6.2.1).





BOEING ANALYSIS CODE LEOTHA

LAMINATE: $(+45/90/-45/0_n/-45/90/+45)$ n = 0,1,2,3 . . .

n = NUMBER OF 0-deg PLIES

PLY THICKNESS = 0.0074 in

MATERIAL: GRAPHITE-EPOXY TAPE LAMINATE

Figure 2.3-1. Buckling of Curved Laminate Fuselage Skin Panels

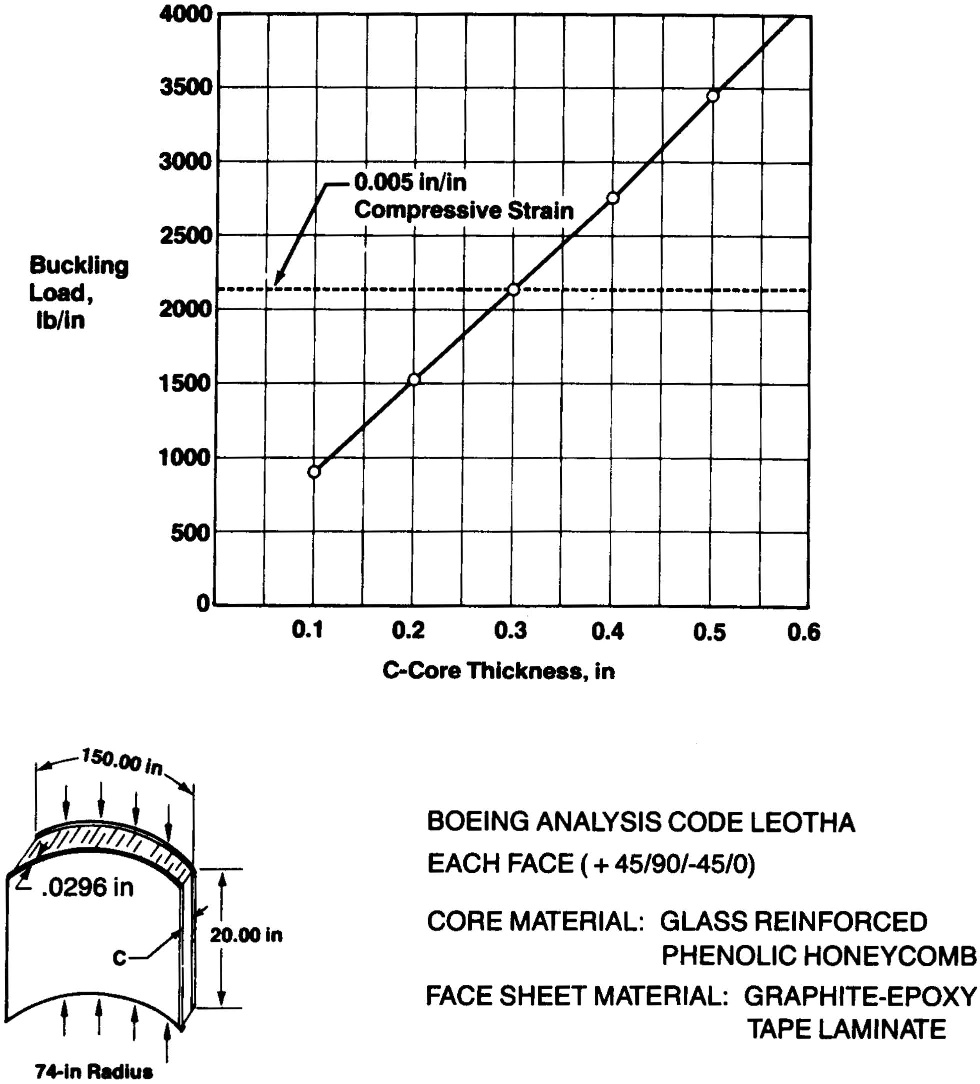


Figure 2.3-2. Buckling of Curved Honeycomb Fuselage Skin Panels

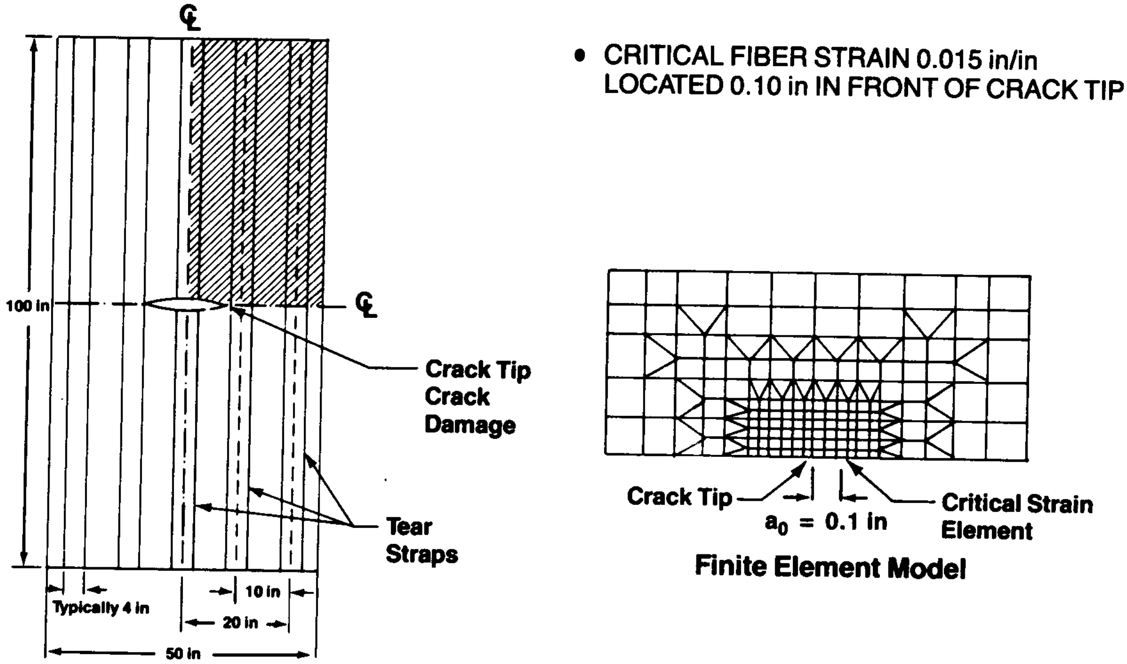


Figure 2.3-3. Tear Strap Fracture Panel Analysis

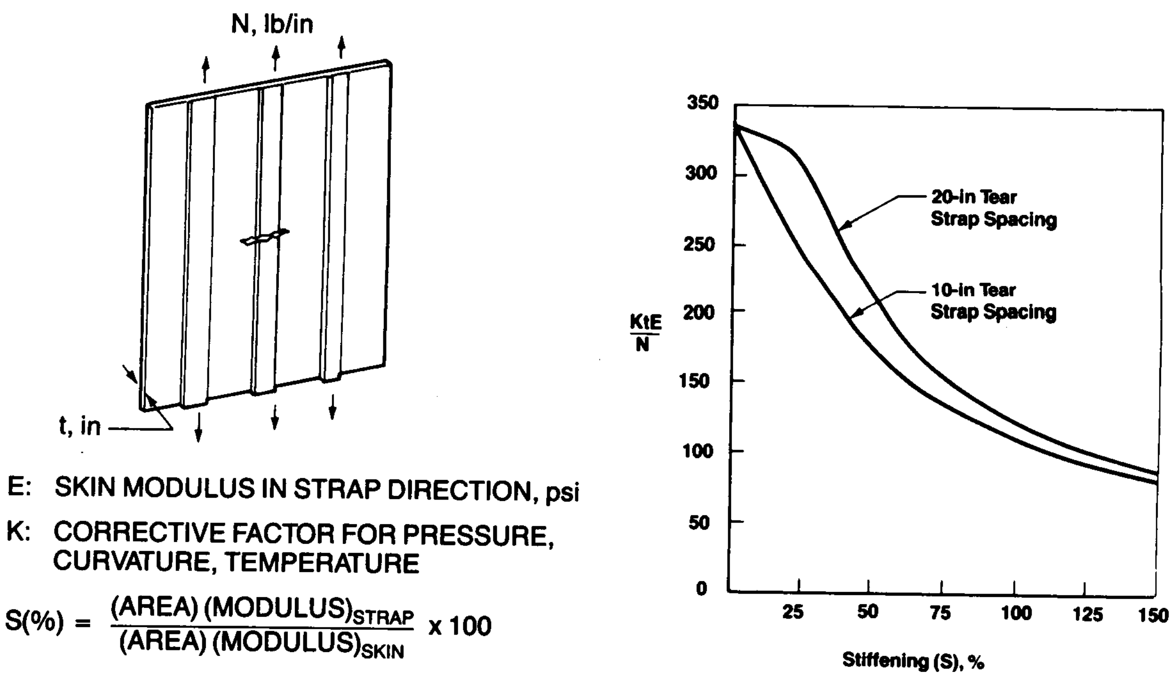


Figure 2.3-4. Tear Strap Design Curves

2.4 BASELINE SECTION

The 757 aft fuselage section was selected as the baseline for design development and for aluminum to composite cost-weight comparisons. The principal characteristics of the 757 airplane are shown in Figure 2.4-1. The baseline study section, shown in Figure 2.4-2, is representative of state-of-the-art, standard body, aluminum fuselage design.

In order to maintain consistency with the current 757, all of the composite concepts retained the same internal and external configuration as the 757 airplane including frame spacing and inner (IML) and outer mold lines (OML). Weight reductions for floor beams, doors, door cutout reinforcement, keel beams and bulkheads were not included in the development of the six composite shell concepts. These components were included when the study section results were extrapolated to a complete fuselage for overall weight reduction estimates (see sec. 3.5).

2.5 DESIGN LOADS

Critical loads in the fuselage generally result from flight conditions that subject the fuselage to positive or negative bending moments, as shown in Figure 2.5-1. The critical loads at particular points in the fuselage study section are shown in Figure 2.5-2. In the crown, the maximum tensile loads result from bending and internal pressure. In the keel, the maximum compression loads result from bending with no internal pressure. The fuselage concepts were sized using the loads shown in Figure 2.5-2.

2.6 CONCEPT DEFINITION

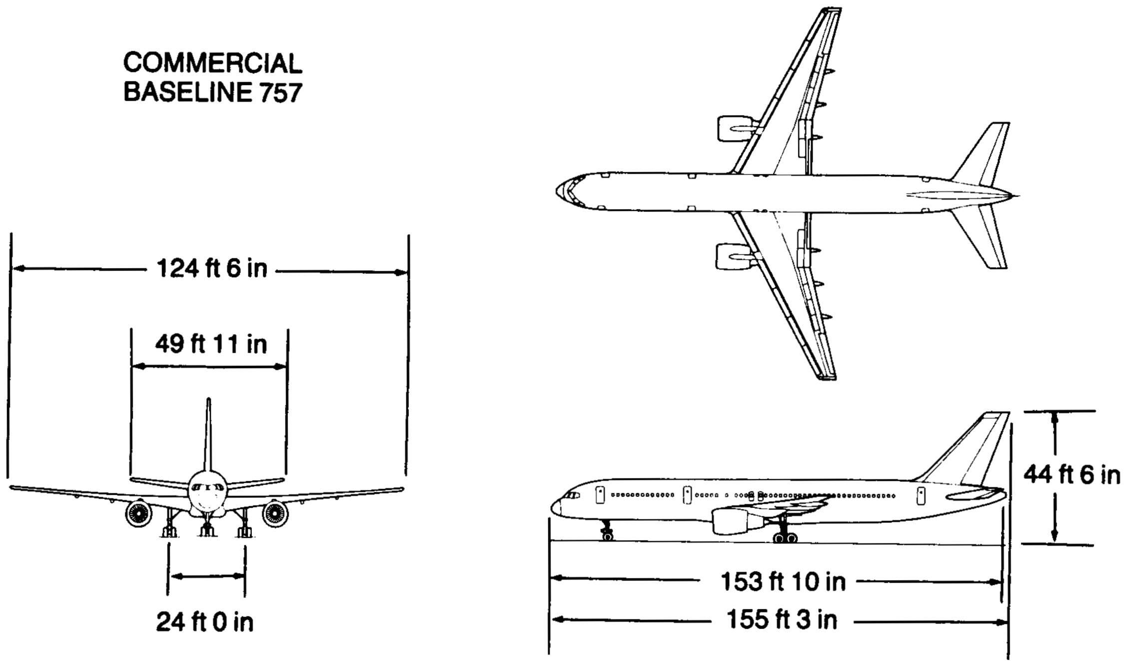
At the start of this program, six fuselage design concepts, shown in Figure 2.6-1, were chosen as having good potential for composite fuselage application. These concepts can be characterized into three groups, as follows:

- Full-depth honeycomb core with laminate face sheets, concept 1
  - Fully stabilized skin
- Laminate skin with discrete stringers, concepts 2, 3 and 4
  - Skin buckling allowed at 30% DUL
- Thin honeycomb core with discrete stringers, concepts 5 and 6
  - Fully stabilized skin

These design concepts have been developed to a level sufficient for comparing structural efficiency, weight, and ease of manufacturability.

The composite shell was designed using three skin panels spliced at the crown and lower sides. These splice locations are shown in a cross-sectional view of the shell, Figure 2.6-2. The design effort concentrated on the shell details, since the skins, stringers, and frames comprise the major portion of the fuselage weight, as shown in Figure 2.2-1. Stringer spacing was selected to provide sufficient space for frame shear ties in the side and keel areas. Stringer spacing in the crown area was selected to provide adequate stiffening for reverse bending buckling requirements. The body frames were sized for overall fuselage stability, as described in Section 2.3.

Damage tolerance for fuselage structures is enhanced by adding extra material to the skin in the form of tear straps. The tear straps are integrated with the skin during fabrication by interleaving 3- to 4-inch wide 0-deg plies into the skin at frame and stiffener locations.



PRINCIPAL CHARACTERISTICS	
MAXIMUM TAXI WEIGHT (BASIC)	221,000 lb
MAXIMUM TAKEOFF WEIGHT	
BASIC	220,000 lb
MAXIMUM LANDING WEIGHT	198,000 lb
MAXIMUM ZERO FUEL WEIGHT	184,000 lb
ENGINE THRUST	37,000 lb
PASSENGER CAPACITY	175-200
FUEL CAPACITY	10,880 gal
CARGO CAPACITY	
ALL BULK	1,700 ft^3
MAXIMUM OPERATION SPEED	
CRUISE AIR SPEED	350 knots
MACH NUMBER	0.86

Figure 2.4-1. Commercial Transport Baseline Model

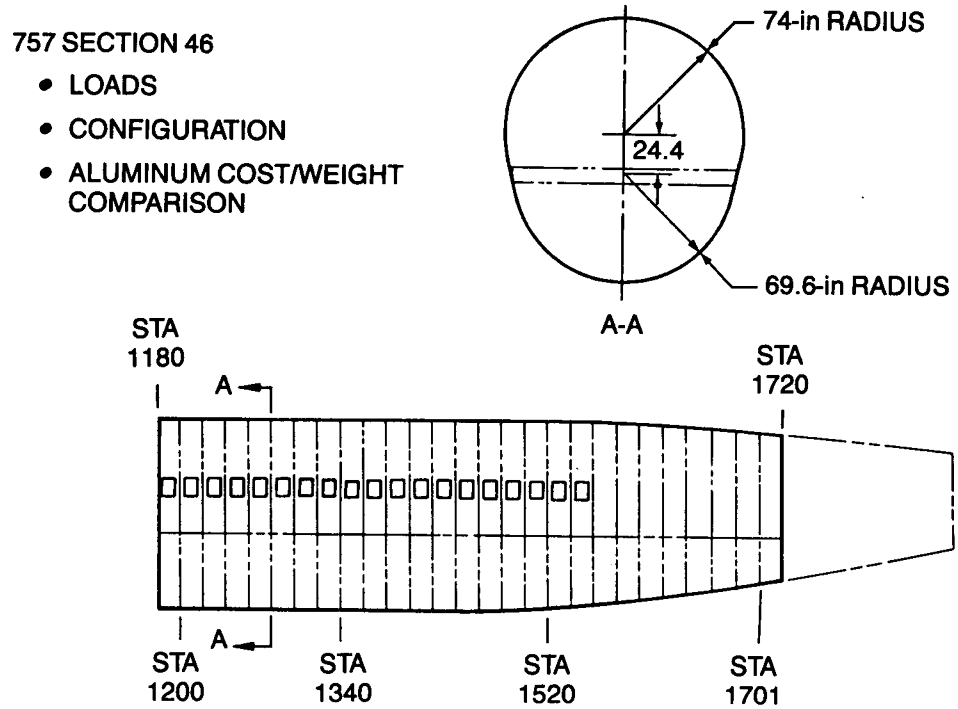


Figure 2.4-2. Commercial Transport Baseline Study Section

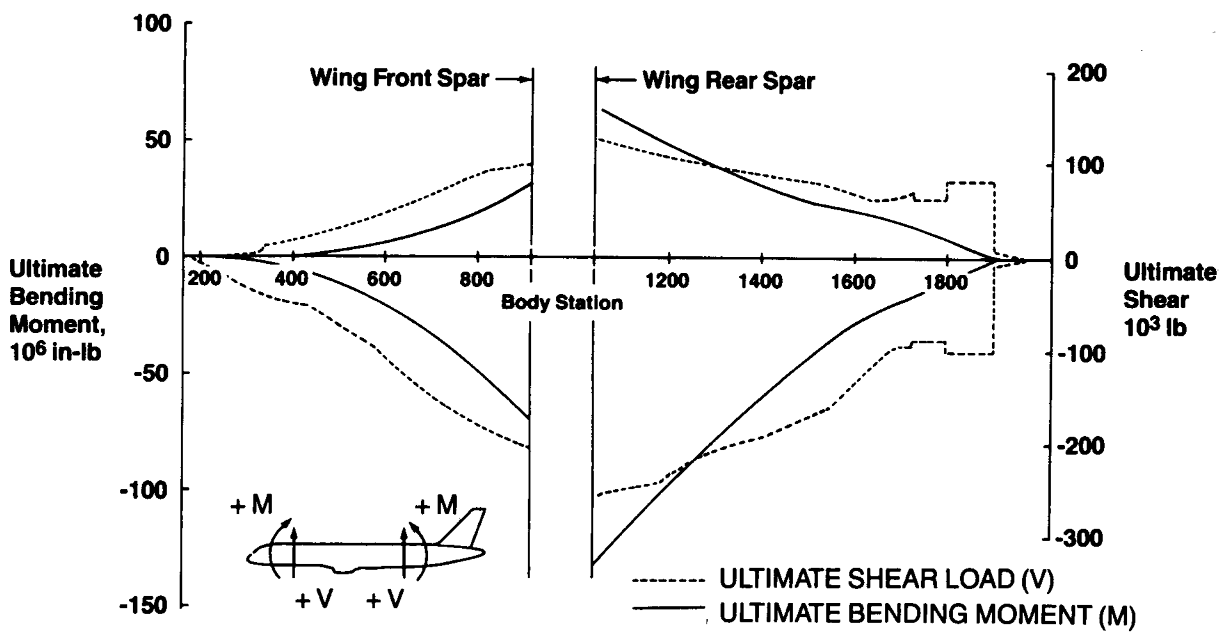


Figure 2.5-1. Ultimate Bending Moment and Shear Load Envelopes for Fuselage During Flight

STATION	CROWN LOAD, lb/in	SIDE PANEL LOAD, lb/in	KEEL PANEL LOAD, lb/in
1200	(TAIL DOWN) 1 N_x = +5000 q = +200 (TAIL UP) 2 N_x = −1800 q = −100	N_x = −500 q = +1100 3	(TAIL DOWN) N_x = −5500 q = +200 4
1340	(TAIL DOWN) N_x = +3670 q = +200 (TAIL UP) N_x = −1330 q = −80	N_x = −500 q = +900 3	(TAIL DOWN) N_x = −3560 q = +200 4
1520	(TAIL DOWN) N_x = +2500 q = +200 (TAIL UP) N_x = −900 q = −50	N_x = +500 q = +650 3	(TAIL DOWN) N_x = −2000 q = +200 4
1701	(TAIL DOWN) N_x = +1950 q = +250 (TAIL UP) N_x = −900 q = +250	N_x = +500 q = +600 3	(TAIL DOWN) N_x = −1500 q = +250 4

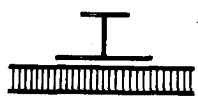
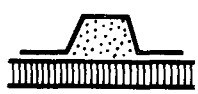
1 TAIL DOWN LOAD, CROWN IN TENSION

2 TAIL UP LOAD, CROWN IN COMPRESSION

3 TAIL UP LOAD PRODUCES LOWER SIDE PANEL SHEAR LOADS

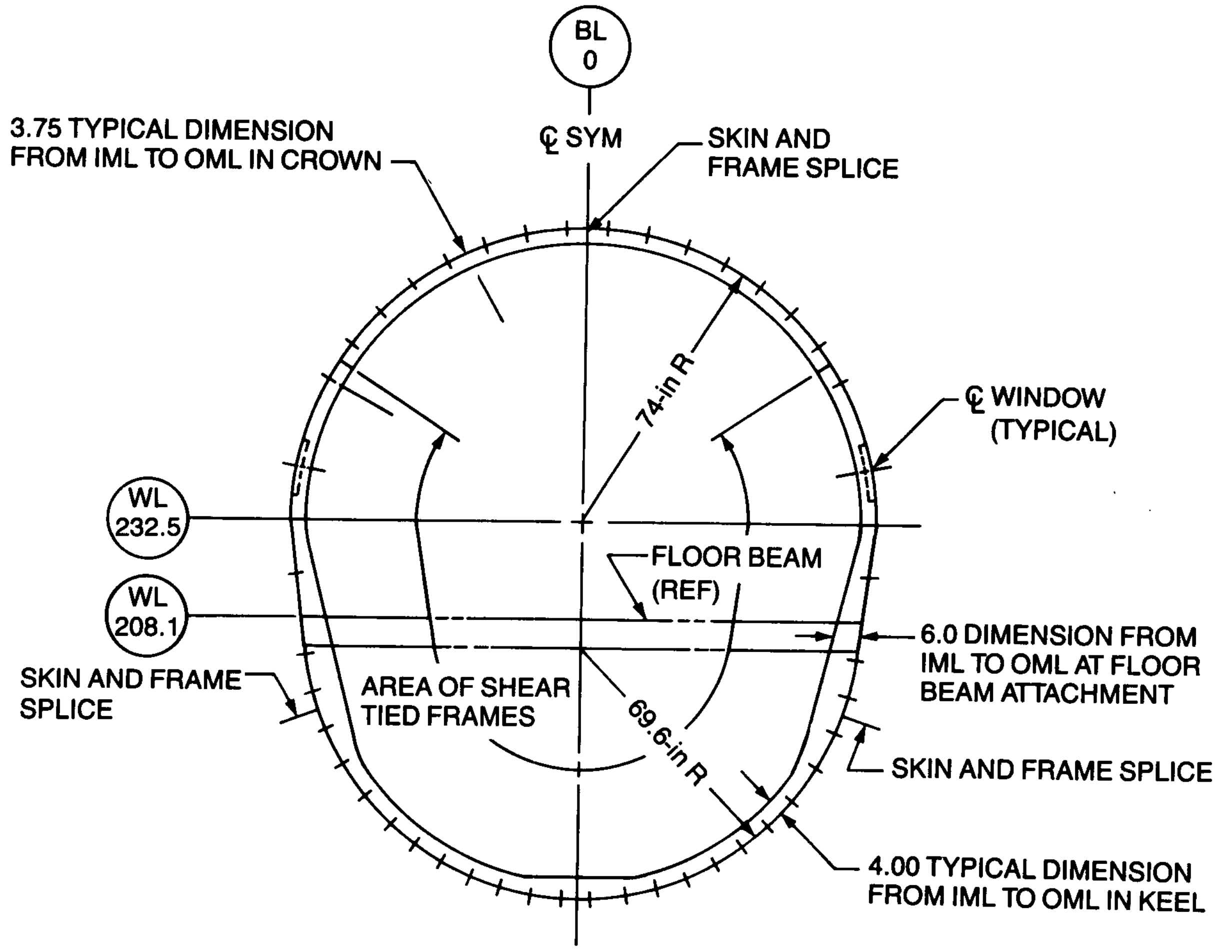
4 TAIL UP LOAD PRODUCES TENSION IN KEEL PANEL, BUT LOAD INTENSITIES DO NOT INFLUENCE DESIGN

Figure 2.5-2. Aftbody Study Section Design Loads

1 	• FULL-DEPTH HONEYCOMB CORE
2 	• LAMINATE SKIN • COCURED I-STRINGERS
3 	• LAMINATE SKIN • COCURED FOAM-FILLED HAT SECTION STRINGERS • BONDED FRAMES
4 	• LAMINATE SKIN • COCURED FOAM-FILLED HAT SECTION STRINGERS • FRAMES MECHANICALLY ATTACHED
5 	• HONEYCOMB CORE • COCURED I-STRINGERS
6 	• HONEYCOMB CORE • COCURED FOAM-FILLED HAT SECTION STRINGERS

20-in FRAME SPACING

Figure 2.6-1. Description of Fuselage Design Concepts



STATION	OUTSIDE SKIN RADIUS, in	
	CROWN	KEEL
1200	74.0	69.6
1340	74.0	69.6
1520	74.0	65.8
1701	65.0	59.9

DIMENSIONS IN INCHES

Figure 2.6-2. Commercial Transport Composite Fuselage Cross Section

2.6.1 Full-Depth Honeycomb Sandwich Skin

The configuration for the full depth honeycomb skin design, Concept 1, is shown in Figure 2.6-3. The skin configuration was designed to meet all requirements of extensional strain and stability without need for stringers. Body frames are mechanically attached to a T-section that is cocured to the honeycomb skin during shell fabrication.

2.6.2 Laminate Skin With Stringers

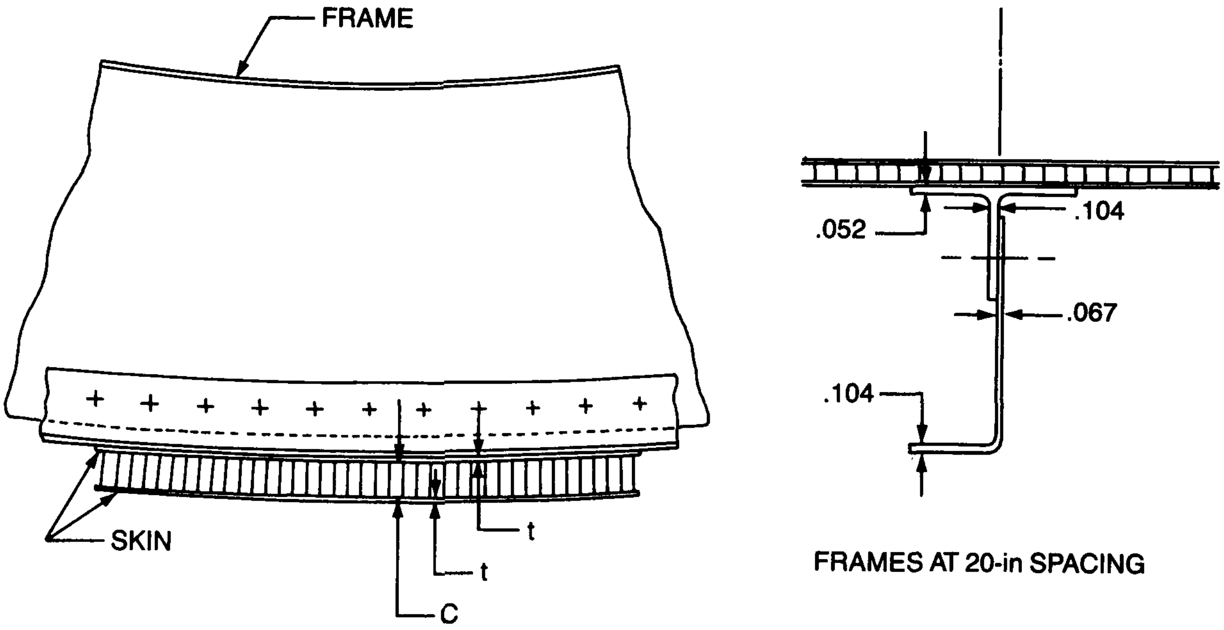
The second group of design configurations consists of either I-section (Concept 2) or hat section stringers (Concepts 3 and 4) cobonded to a laminate skin. The configuration of the skin and stringer in the I-section stiffened laminate skin design, Concept 2, are shown in Figure 2.6-4. In order to carry a majority of the axial loading and to create an efficient stringer section for column stability, the I-section stringers were designed with a high number of 0-deg plies in the cap, oriented along the length of the stringer. The skins were sized to be stable up to 30% of design ultimate load (DUL) using cross ply laminates containing a high percentage of layers oriented at 45 deg to the extensional load direction. The frame for Concept 2, shown in Figure 2.6-5, is mechanically attached to the stringer flange and to the skin using shear ties in the side and keel region. In the crown region, the frame is connected to the outer shell by mechanically attaching the frame to the stringer flanges only.

The stringer configurations for the hat section stiffened laminate designs, Concepts 3 and 4, are shown in Figure 2.6-6. The hat section stringer is laid up over the foam core and cocured to the skin. In addition to facilitating fabrication, as discussed in Section 5.0, the foam core provides lateral stability to the stringer webs and flange. The hat stringer has a substantially wider base than the I-stringer. This reduces the skin thickness requirements by narrowing the width of the skin susceptible to buckling. Because of this, the skins for Concepts 3 and 4, shown in Figure 2.6-6, are thinner than the skins of the I-stiffened laminate designs of Figure 2.6-4.

The frame configurations for Concepts 3 and 4 are shown in Figure 2.6-7. In the crown region, a Z-section frame is attached mechanically to the shell using a T-cross section stringer clip that is machined to provide clearance over the hat stringer. An alternative attachment method is shown that mechanically attaches the frame in the crown directly through the cap and core of the stringer. The difference between concepts 3 and 4 is that each concept uses a different frame design in the keel. In Concept 3, a Z-section frame is mechanically attached to the skin via a T-section that is cocured to the skin. In Concept 4, a channel-section frame is mechanically attached directly to the skin. A fail safe angle is cobonded to the frame. The angle, together with the inside part of the channel, provides the necessary frame stiffness and damage tolerance.

2.6.3 Honeycomb Skin With Stringers

The remaining designs consist of either I-section (Concept 5) or hat section stringers (Concept 6) cobonded to a honeycomb stabilized skin. The I-stringer configuration, shown in Figure 2.6-8, is similar to the configuration of the I-stringer in the laminate skin design, Figure 2.6-4. Since the honeycomb core stabilizes the skin, less laminate material is needed in the skin of the honeycomb design. The frame for Concept 5, shown in Figure 2.6-9, is mechanically attached to the flanges of the I-stringer in the crown region. In the keel region the frame is shear tied to the skin between stringers. The hat stringer configuration on honeycomb skin, shown in Figure 2.6-10, is similar to the configurations of the hat section stringer on laminate skin. Frame attachment details are shown in Figure 2.6-11. In the crown, potting inserts in the skin are used to provide hard points suitable for mechanically attaching the tension clips to the skin.



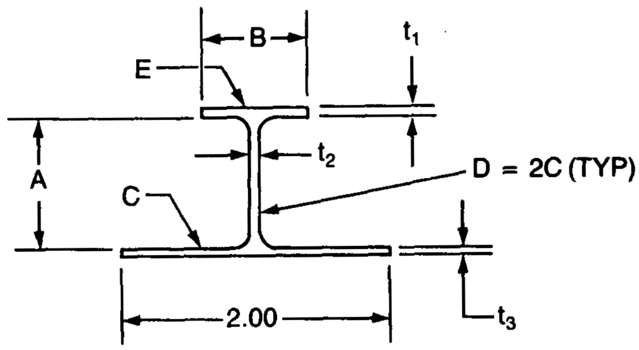
DIMENSIONS IN INCHES

STATION	CROWN			KEEL		
	C, in	t, in	LAYUP	C, in	t, in	LAYUP
1200	.20	.052	90/0/45/0/-45/0/90	.60	.067	$(0/-45/90/45/\bar{0})_s$
1340	.20	.037	0/-45/90/45/0	.50	.052	90/0/45/0/-45/0/90
1520	.15	.037	0/45/90/45/0	.35	.037	0/-45/90/45/0
1701	.15	.037	0/-45/90/45/0	.30	.037	0/-45/90/45/0

Figure 2.6-3. Skin and Frame Configurations of Full-Depth Honeycomb Skin Design, Concept 1

STA	SKIN — CROWN — LAYUP	t
1200	$+45/90/-45/+45/-45/0_3/-45/+45/-45/90/+45$	.0962
1340	$+45/90/-45/+45/-45/0_2/-45/+45/-45/90/+45$	.0888
1520	$+45/90/-45/0_4/-45/90/+45$	.0740
1701	$+45/90/-45/0_4/-45/90/+45$	.0740

STA	SKIN — KEEL — LAYUP	t
1200	$+45/90/-45/0_2/+45/-45/0_2/-45/+45/0_2/-45/90/+45$	.1184
1340	$+45/90/-45/+45/-45/0_3/-45/+45/-45/90/+45$	.0962
1520	$+45/90/-45/0_5/-45/90/+45$	.0814
1701	$+45/90/-45/0_2/-45/90/+45$	.0592



STRINGER SPACING
CROWN 10.0 in
KEEL 8.0 in

DIMENSIONS IN INCHES

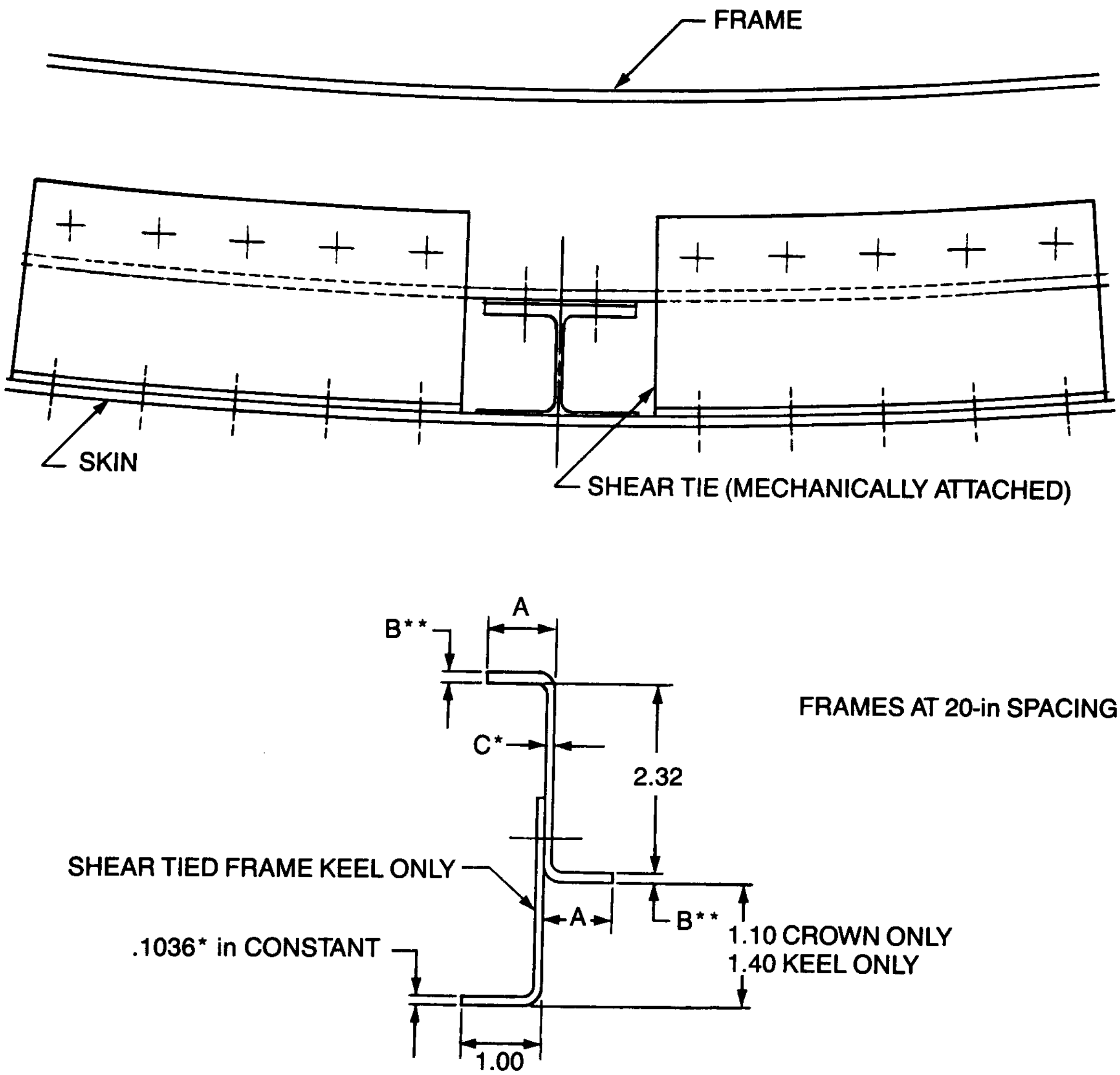
STRINGER CONFIGURATION

STA	CROWN A	CROWN B	CROWN t_1	CROWN t_2	CROWN t_3	KEEL A	KEEL B	KEEL t_1	KEEL t_2	KEEL t_3
1200	1.00	.80	.0814	.0592	.0296	1.17	1.30	.1332	.0740	.0370
1340	1.00	.80	.0814	.0592	.0296	1.17	1.20	.1184	.0740	.0370
1520	1.00	.60	.0592	.0592	.0296	1.17	.74	.0740	.0740	.0370
1701	1.00	.50	.0592	.0592	.0296	1.17	.50	.0592	.0740	.0370

STA	CROWN C	CROWN E
1200	$+45/90/-45/0$	$+45/90/-45/0_5/-45/90/+45$
1340	$+45/90/-45/0$	$+45/90/-45/0_5/-45/90/+45$
1520	$+45/90/-45/0$	$+45/90/-45/0_2/-45/90/+45$
1701	$+45/90/-45/0$	$+45/90/-45/0_2/-45/90/+45$

STA	KEEL C	KEEL E
1200	$+45/90/-45/0_2$	$+45/90/-45/0_5/+45/-45/0_5/-45/90/+45$
1340	$+45/90/-45/0_2$	$+45/90/-45/0_4/+45/-45/0_4/-45/90/+45$
1520	$+45/90/-45/0_2$	$+45/90/-45/0_4/-45/90/+45$
1701	$+45/90/-45/0_2$	$+45/90/-45/0_2/-45/90/+45$

Figure 2.6-4. Laminate Skin and I-Section Stringer Configurations for Concept 2

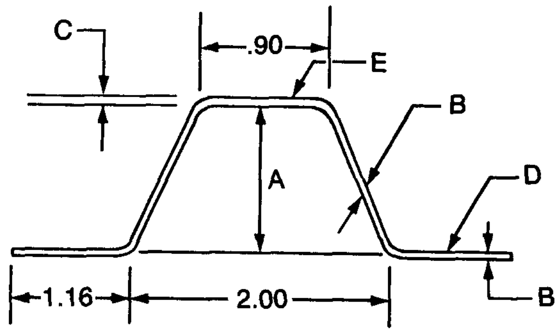


FRAME CONFIGURATION						
STA	CROWN			KEEL		
	A	B**	C*	A	B**	C*
1200	.90	.1924	.1184	.85	.1258	.0962
1340	.80	.1480	.1036	.85	.1110	.0666
1520	.80	.0962	.0666	.85	.1110	.0666
1701	.80	.0962	.0666	.85	.1110	.0666

* 100% FABRIC PLIES
** C PLUS 0° TAPE PLIES

DIMENSIONS IN INCHES

Figure 2.6-5. Frame Configuration in I-Stiffened Laminate Skin Design, Concept 2



STRINGER SPACING
CROWN 10.0 in
KEEL 8.0 in

SKIN		
STA	CROWN	
	LAYUP	t
1200	$+45/90/-45/0_5/-45/90/+45$	.0814
1340	$+45/90/-45/0_4/-45/90/+45$	.0740
1520	$+45/90/-45/0_2/-45/90/+45$	.0592
1701	$+45/90/-45/0_2/-45/90/+45$	.0592
STA	KEEL	
	LAYUP	t
1200	$+45/90/-45/0_5/-45/90/+45$	.0814
1340	$+45/90/-45/0_4/-45/90/+45$	.0740
1520	$+45/90/-45/0/-45/90/+45$	.0518
1701	$+45/90/-45_2/90/+45$	.0444

STRINGER CONFIGURATION						
STA	CROWN			KEEL		
	A	B	C	A	B	C
1200	.95	.0296	.0592	1.12	.0666	.0666
1340	.95	.0296	.0444	1.12	.0370	.0592
1520	.95	.0296	.0370	1.12	.0370	.0370
1701	.95	.0296	.0370	1.12	.0296	.0296

STA	CROWN		KEEL	
	D	E	D	E
1200	$+45/90/-45/0$	$+45/90/-45/0_5$	$+45/90/-45/0_6$	$+45/90/-45/0_6$
1340	$+45/90/-45/0$	$+45/90/-45/0_5$	$+45/90/-45/0_2$	$+45/90/-45/0_5$
1520	$+45/90/-45/0$	$+45/90/-45/0_2$	$+45/90/-45/0_2$	$+45/90/-45/0_2$
1701	$+45/90/-45/0$	$+45/90/-45/0_2$	$+45/90/-45/0$	$+45/90/-45/C$

DIMENSIONS IN INCHES

Figure 2.6-6. Hat Section Stringer Configuration for Laminate Skin Design Concepts 3 and 4

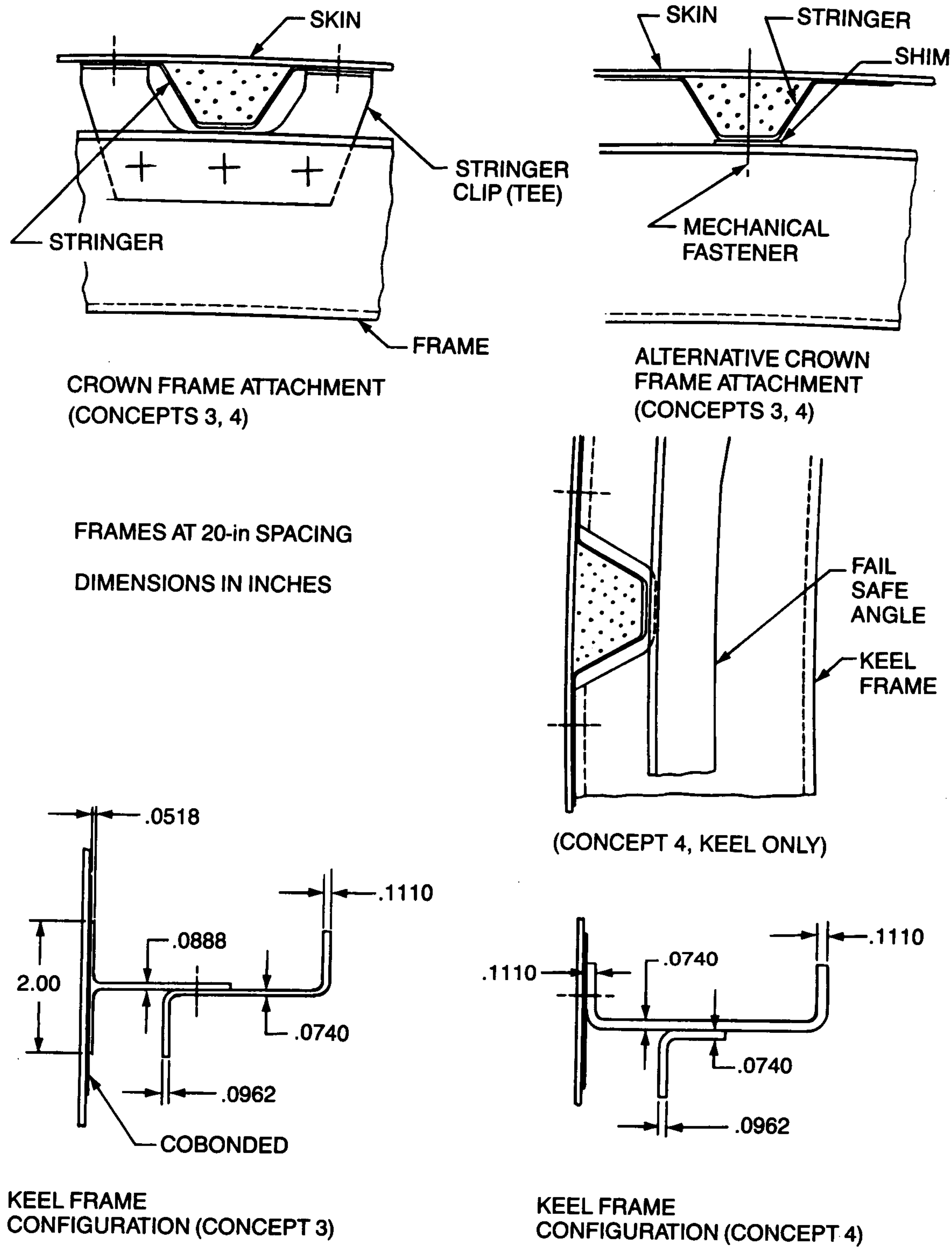
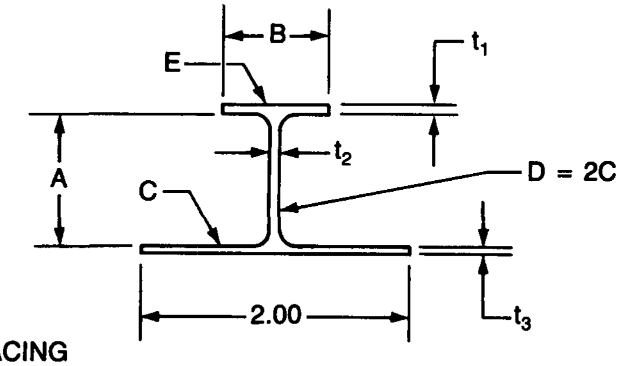
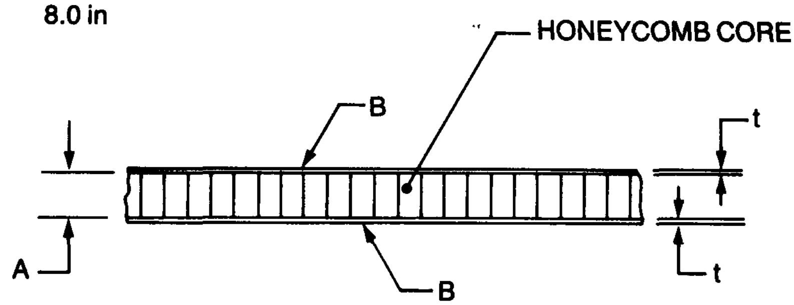


Figure 2.6-7. Frame Configuration in Hat Stiffened Laminate Skin Designs, Concepts 3 and 4



STRINGER SPACING
CROWN 10.0 in
KEEL 8.0 in

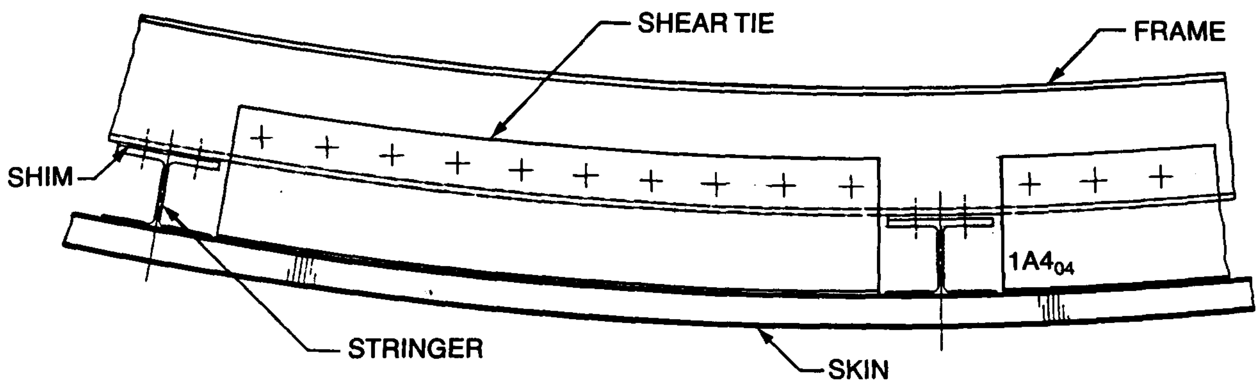


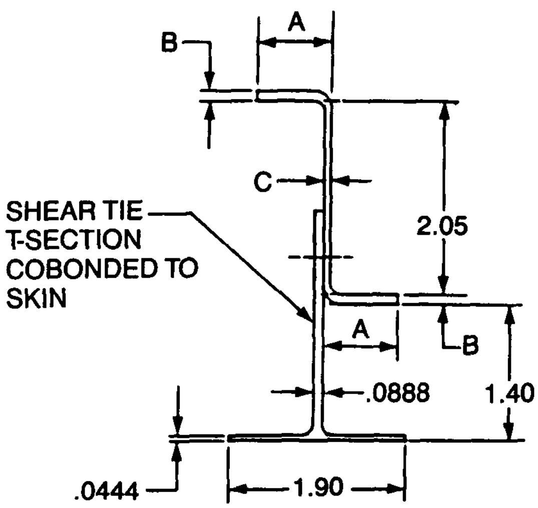
SKIN CONFIGURATION						
	CROWN			KEEL		
STA	A	B	t	A	B	t
1200	.25	+45/90/-45/0_2	.0370	.50	+45/90/-45/0_3	.0444
1340	.20	+45/90/-45/0_2	.0370	.40	+45/90/-45/0_2	.0370
1520	.20	+45/90/-45/0	.0296	.30	+45/90/-45/0	.0296
1701	.20	+45/90/-45/0	.0296	.20	+45/90/-45/0	.0296

STRINGER CONFIGURATION							
	CROWN						
STA	A	B	t_1	t_2	t_3	C	E
1200	1.00	1.20	.0814	.0592	.0296	+45/90/-45/0	+45/90/-45/0_5/-45/90/+45
1340	1.00	.80	.0814	.0592	.0296	+45/90/-45/0	+45/90/-45/0_5/-45/90/+45
1520	1.00	.60	.0666	.0592	0296	+45/90/-45/0	+45/90/-45/0_3/-45/90/+45
1701	1.00	.50	.0666	.0592	.0296	+45/90/-45/0	+45/90/-45/0_3/-45/90/+45
	KEEL						
STA	A	B	t_1	t_2	t_3	C	E
1200	1.17	1.30	.1332	.0740	.0370	+45/90/-45/0_2	+45/90/-45/0_5/+45/-45/0_5/-45/90/+45
1340	1.09	.90	.1184	.0740	.0370	+45/90/-45/0_2	+45/90/-45/0_4/+45/-45/0_4/-45/90/+45
1520	1.00	.70	0740	.0740	.0370	+45/90/-45/0_7	+45/90/-45/0_4/-45/90/+45
1701	.90	.70	.0740	.0740	.0370	+45/90/-45/0_2	+45/90/-45/0_4/-45/90/+45

DIMENSIONS IN INCHES

Figure 2.6-8. Honeycomb Skin and I-Section Stringer Configurations for Concept 5

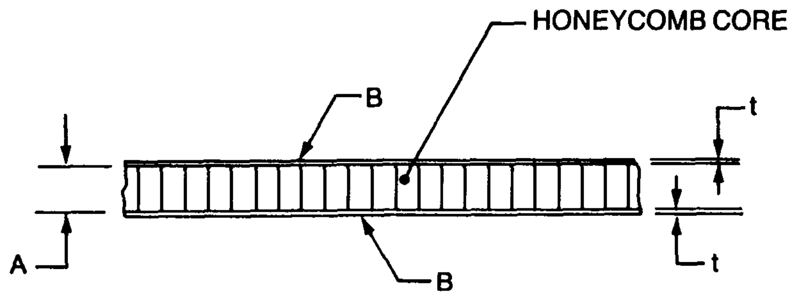


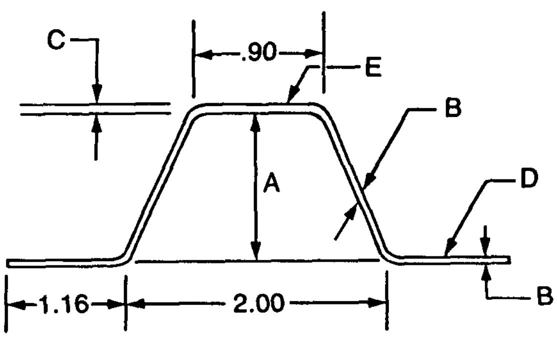


FRAME CONFIGURATION						
STA	CROWN			KEEL		
	A	B	C	A	B	C
1200	.80	.0962	.0666	.85	.1110	.0666
1340	.80	.0962	.0666	.85	.1110	.0666
1520	.80	.0962	.0666	.85	.1110	.0666
1701	.80	.0962	.0666	.85	.1110	.0666

DIMENSIONS IN INCHES

Figure 2.6-9. Frame Configuration in I-Stiffened Honeycomb Skin Design, Concept 5





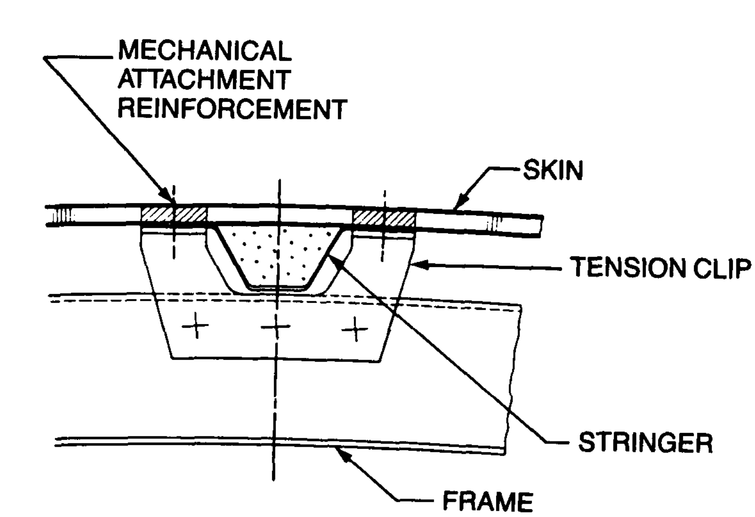
STRINGER SPACING
CROWN 10.0 in
KEEL 8.0 in

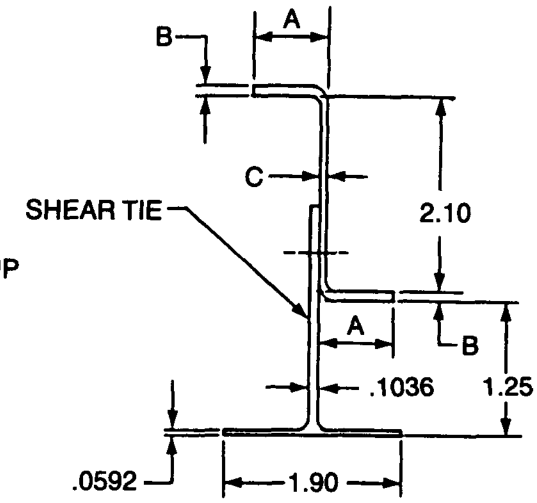
STRINGER CONFIGURATION					
STA	CROWN				
	A	B	C	D	E
1200	.95	.0296	.0592	$+45/90/-45/0$	$+45/90/-45/0_5$
1340	.95	.0296	.0444	$+45/90/-45/0$	$+45/90/-45/0_3$
1520	.95	.0296	.0370	$+45/90/-45/0$	$+45/90/-45/0_2$
1701	95	.0296	.0370	$+45/90/-45/0$	$+45/90/-45/0_2$
	KEEL				
	A	B	C	D	E
1200	1.12	.0592	.0666	$+45/90/-45/0_5$	$+45/90/-45/0_6$
1340	1.12	.0370	.0592	$+45/90/-45/0_2$	$+45/90/-45/0_5$
1520	1.12	.0370	.0370	$+45/90/-45/0_2$	$+45/90/-45/0_2$
1701	1.12	.0296	.0370	$+45/90/-45/0$	$+45/90/-45/0_2$

SKIN CONFIGURATION						
STA	CROWN			KEEL		
	A	B	t	A	B	t
1200	.25	$+45/90/-45/0_2$	.0370	.50	$+45/90/-45/0_3$	.0444
1340	.20	$+45/90/-45/0_2$	.0370	.40	$+45/90/-45/0_2$	.0370
1520	.20	$+45/90/-45/0$	.0296	.30	$+45/90/-45/0$	.0296
1701	.20	$+45/90/-45/0$	.0296	.20	$+45/90/-45/0$	.0296

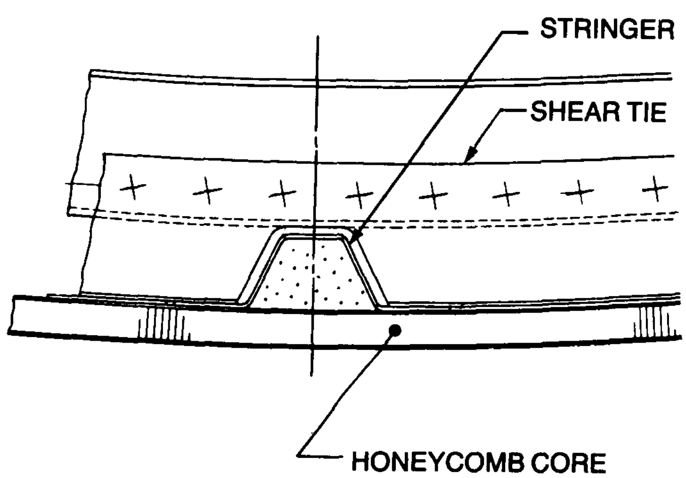
DIMENSIONS IN INCHES

Figure 2.6-10. Honeycomb Skin and Hat Section Stringer Configurations for Concept 6





FRAMES AT 20-in SPACING



FRAME CONFIGURATION						
STA	CROWN			KEEL		
	A	B	C	A	B	C
1200	.80	.0962	.0666	.85	.1110	.0666
1340	↓	↓	↓	↓	↓	↓
1520	↓	↓	↓	↓	↓	↓
1701	↓	↓	↓	↓	↓	↓

DIMENSIONS IN INCHES

Figure 2.6-11. Frame Configuration in Hat Stiffened Honeycomb Skin Design, Concept 6

2.6.4 Window Frames

Window frame concepts that can be used in laminate and honeycomb skins are shown in Figures 2.6-12 and 2.6-13. The frame concepts shown in Figure 2.6-12 consist of graphite plies wrapped around a foam core. The concepts shown in Figure 2.6-13 could be made from graphite-epoxy molded fabric and tape. The skin in the window area has been increased in thickness to reduce the load concentration effects around the cutout. The window frame provides torsional stiffness to the window cutout edge to redistribute the window pane pressure and to provide out-of-plane stiffness around the edge of the cutout.

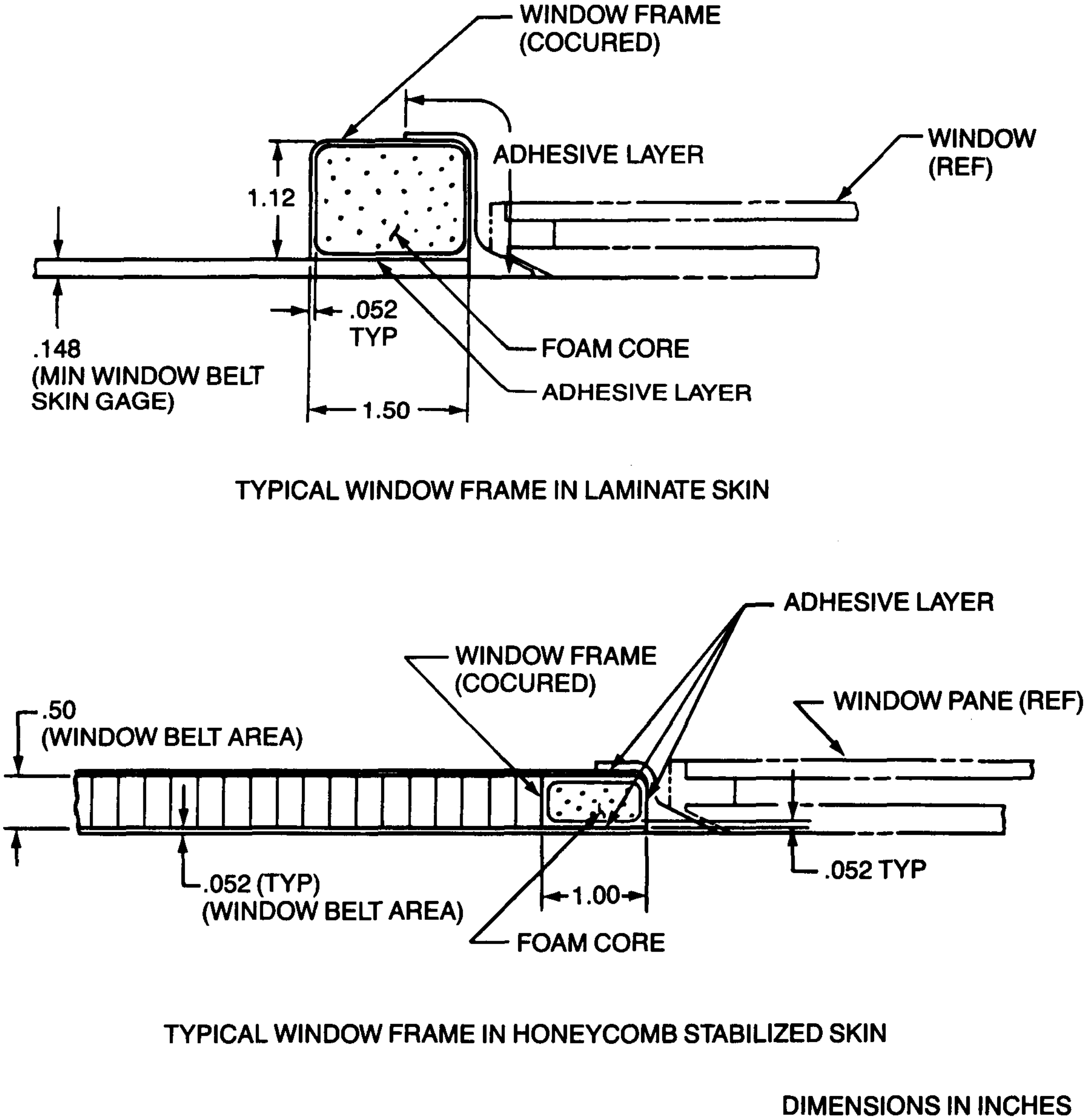


Figure 2.6-12. Foam Filled Window Frame Designs in Laminate and Honeycomb Skins

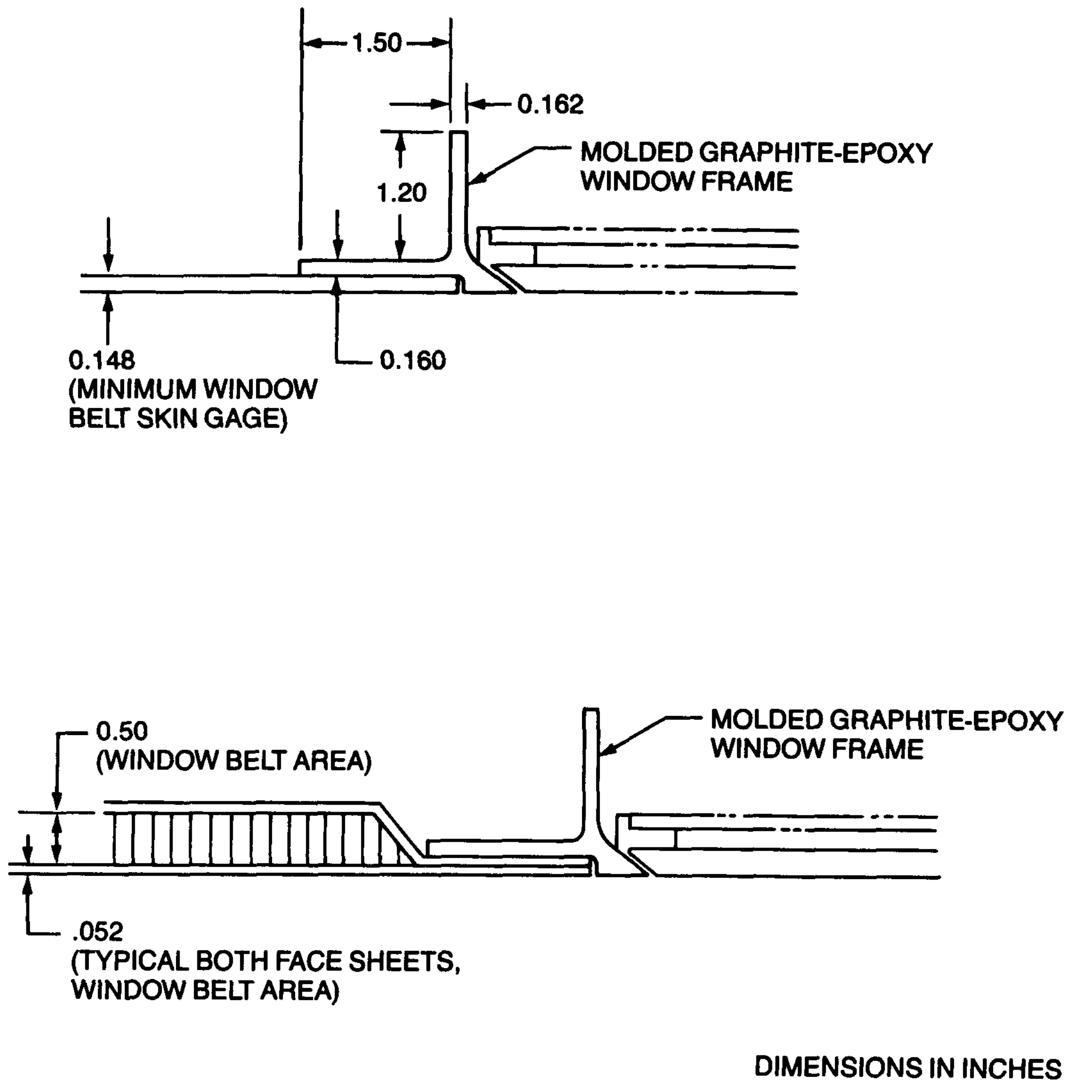


Figure 2.6-13. Alternate Window Frame Designs for Laminate and Honeycomb Skins

3.0 CONCEPT EVALUATION

3.1 DESIGN STRAINS

Each of the six selected design concepts have been evaluated to ensure that the requirements for strength, stability, and damage tolerance have been met without exceeding strain allowables. The strains at the ultimate design loads for each of the six concepts are summarized in Figures 3.1-1 through 3.1-5. These strain values are derived from the axial design loads and the extensional stiffness of the section. For laminate skin panels loaded in tension, and honeycomb panels loaded under any conditions, the skin is considered fully effective. When laminate skin panels are loaded in compression, however, only the effective amount of unbuckled skin is included.

The load levels in the fuselage study section are greatest near the wing, and progressively decrease moving aft (figs. 2.5-1, 2.5-2). The design strains also decrease along the length of the study section, indicating that the skin and stringer stiffnesses are not completely tailored to the design loads. The skin and stringer stiffnesses do not vary significantly along the shell because the designs were developed for ready utilization of automated fabrication techniques and the need to meet design criteria and guidelines described in Section 2.1. The stringer heights are kept constant along the length of the fuselage to simplify their construction. In addition, the amount that the gages of the skin and stringers could be changed along the length of the fuselage was controlled by laminate constraints of symmetry, balance, modulus, and per-ply-thickness. As design and manufacturing technology are further developed, greater optimization and further weight reduction can be accomplished.

3.2 WEIGHT COMPARISONS

Itemized weight comparisons of the six selected graphite-epoxy design concepts, described in Section 2.8, to the baseline aluminum design of the fuselage study section (fig. 2.4-1) are shown in Figure 3.2-1. Data used to compile the itemized weight breakdown for the aluminum section is typical of an advanced technology, single aisle, pressurized body section of a medium range Boeing aircraft, modified to represent the study section definition.

The graphite-epoxy material used in each concept is unidirectional tape preimpregnated with 35% resin by weight, resulting in a nominal weight of 0.060 pounds per square foot and nominal thickness per ply of 0.0074 inch. The weight of each concept component was calculated using a ply-by-ply area method using the material gage tables shown on the concept layout drawings for the skins, stringers, and frames. Installation and assembly fastener weights for each component have been included in the component weights.

3.3 COST COMPARISONS

The producibility of the concepts was evaluated in terms of recurring factory labor requirements, shown in Figure 3.3-1. A constant section of fuselage with frames at 20-inch spacing was used to develop relative labor hours. These labor hours assume that current fabrication and inspection methods, discussed in Section 5.0, can be used for all concepts. Accordingly, the concepts that are least labor intensive are the honeycomb sandwich skin concept with no stringers, and the laminate skin concepts with discrete stringers.

Due to the similarities between Concepts 3 and 4, a separate cost evaluation was not made for Concept 4. The fabrication complexities involved with a honeycomb skin in Concepts 5 and 6 create higher labor requirements than similar discrete-stringer-designs with laminate skins. Since frames cannot be efficiently attached directly to a hat section stringer, the frames in Concepts 3, 4, and 6 are attached to the skin. With a honeycomb skin, potting material needs to be inserted into the core to provide solid attachment points for the body frames. The insertion of potting into the skin is time consuming, as indicated by the high labor requirements of Concept 6 (see fig. 2.6-11). The labor penalty associated with Concept 5 is not as severe since the crown frames can be attached directly to the flanges of the I-section stringer, therefore eliminating the need for extra potting in the honeycomb skin.

Further discussion on the manufacturing evaluation of the design concepts is provided in Section 5.2.

STATION	TENSION [1]		COMPRESSION [2]		
	DESIGN LOAD N, lb/in	STRAIN AT DESIGN LOAD, in/in	DESIGN LOAD, N, lb/in	STRAIN AT DESIGN LOAD, in/in	SKIN BUCKLING LOAD, lb/in [3]
KEEL					
1200	--	--	-5500	-.0048	-7960
1340	--	--	-3560	-.0036	-4870
1520	[4] --	[4] --	-2000	-.0029	-2790
1701	--	--	-1500	-.0022	-2270
CROWN					
1200	5000	.0050	-1800	-.0018	-2100
1340	3670	.0053	-1330	-.0019	-1550
1520	2500	.0036	- 900	-.0013	-1250
1701	1950	.0028	- 900	-.0013	-1430

[1] ALLOWABLE TENSION STRAIN = 0.006 in/in

[2] ALLOWABLE COMPRESSION STRAIN = 0.005 in/in

[3] BOEING ANALYSIS CODE LEOTHA

[4] TENSION LOADING IN KEEL DOES NOT INFLUENCE DESIGN

Figure 3.1-1. Analysis of Unstiffened Honeycomb Design (Concept 1)

STATION	STRINGER SPACING in	SMEARED THICKNESS t, in	TENSION [1]		COMPRESSION [2]				
			DESIGN LOAD N, lb/in	STRAIN AT DESIGN LOAD, in/in	DESIGN LOAD N, lb/in	STRAIN AT DESIGN LOAD, in/in	EULER BUCKLING LOAD, lb/in	LOAD IN SKIN AT 30% DUL, lb/in	SKIN BUCKLING ALLOWABLE, lb/in
KEEL									
1200	8.0	.166	—	—	−5500	−.0042	−9560	−1170	−1390
1340	8.0	.140	— [3]	— [3]	−3560	−.0038	−6990	− 700	− 810
1520	7.6	.167	—	—	−2000	−.0025	−3500	− 420	− 550
1701	6.8	.094	—	—	−1500	−.0026	−2200	− 250	− 290
CROWN									
1200	10.4	.119	5000	.0046	−1800	−.0031	−2050	− 440	− 550
1340	10.4	.111	3670	.0038	−1330	−.0026	−1970	− 310	− 460
1520	10.4	.094	2500	.0029	− 900	−.0021	−1020	− 220	− 280
1701	9.8	.094	1950	.0022	− 900	−.0020	− 980	− 220	− 340

[1] ALLOWABLE TENSION STRAIN = 0.006 in/in
[2] ALLOWABLE COMPRESSION STRAIN = −0.005 in/in
[3] TENSION LOADING IN KEEL DOES NOT INFLUENCE DESIGN

Figure 3.1-2. Analysis of I-Section Stiffened Laminate Skin Design (Concept 2)

STATION	STRINGER SPACING in	SMEARED THICKNESS t, in	TENSION [1] DESIGN LOAD N, lb/in	TENSION [1] STRAIN AT DESIGN LOAD, in/in	COMPRESSION [2] DESIGN LOAD N, lb/in	COMPRESSION [2] STRAIN AT DESIGN LOAD, in/in	COMPRESSION [2] EULER BUCKLING LOAD, lb/in	COMPRESSION [2] LOAD IN SKIN AT 30% DUL, lb/in	COMPRESSION [2] SKIN BUCKLING ALLOWABLE, lb/in
KEEL									
1200	8.0	.140	—	—	−5500	−.0038	−5960	−840	−1080
1340	8.0	.115	— [3]	— [3]	−3560	−.0035	−3790	−640	−820
1520	7.6	.092	—	—	−2000	−.0031	−2310	−250	−350
1701	6.9	.084	—	—	−1500	−.0029	−1750	−140	−360
CROWN									
1200	10.4	.109	5000	.0045	−1800	−.0022	−1960	−400	−470
1340	10.4	.101	3670	.0038	−1330	−.0019	−1720	−290	−360
1520	10.4	.085	2500	.0037	−900	−.0018	−1050	−170	−210
1701	9.8	.086	1950	.0028	−900	−.0018	−960	−180	−250

[1] ALLOWABLE TENSION STRAIN = 0.006 in/in
[2] ALLOWABLE COMPRESSION STRAIN = −0.005 in/in
[3] TENSION LOADING IN KEEL DOES NOT INFLUENCE DESIGN

Figure 3.1-3. Analysis of Hat Section Stiffened Laminate Skin Design (Concepts 3 and 4)

STATION	STRINGER SPACING in	[3] SMEARED THICKNESS t, in	TENSION [1] DESIGN LOAD N, lb/in	TENSION [1] STRAIN AT DESIGN LOAD, in/in	COMPRESSION [2] DESIGN LOAD N, lb/in	COMPRESSION [2] STRAIN AT DESIGN LOAD, in/in	COMPRESSION [2] EULER BUCKLING LOAD, lb/in	COMPRESSION [2] LOAD IN SKIN AT 30% DUL, lb/in	COMPRESSION [2] SKIN BUCKLING ALLOWABLE, lb/in
KEEL									
1200	14.4	.112	—	—	−5500	−.0046	−9830	−4470	−7200
1340	14.4	.095	— [4]	— [4]	−3560	−.0040	−5220	−2770	−4710
1520	13.6	.075	—	—	−2000	−.0035	−2080	−1490	−2850
1701	12.4	.075	—	—	−1500	−.0026	−1480	−1130	−1890
CROWN									
1200	14.0	.092	5000	.0059	−1800	−.0021	−3160	−1490	−2620
1340	14.0	.089	3670	.0045	−1330	−.0016	−2220	−1130	−1990
1520	14.0	.073	2500	.0047	− 900	−.0017	−1300	− 730	−1600
1701	12.4	.072	1950	.0037	− 900	−.0017	−1170	− 730	−1890

[1] ALLOWABLE TENSION STRAIN = 0.006 in/in
[2] ALLOWABLE COMPRESSION STRAIN = −0.005 in/in
[3] SMEARED THICKNESS DOES NOT INCLUDE HONEYCOMB MATERIAL
[4] TENSION LOADING IN KEEL DOES NOT INFLUENCE DESIGN

Figure 3.1-4. Analysis of I-Section Stiffened Honeycomb Skin Design (Concept 5)

STATION	STRINGER SPACING in	[3] SMEARED THICKNESS t, in	TENSION [1]		COMPRESSION [2]				
			DESIGN LOAD N, lb/in	STRAIN AT DESIGN LOAD, in/in	DESIGN LOAD N, lb/in	STRAIN AT DESIGN LOAD, in/in	EULER BUCKLING LOAD, lb/in	LOAD IN SKIN AT 30% DUL, lb/in	SKIN BUCKLING ALLOWABLE, lb/in
KEEL									
1200	14.4	.119	—	—	−5500	−.0042	−7260	−4040	−8460
1340	14.4	.097	[4]	[4]	−3560	−.0039	−4080	−2750	−5660
1520	13.6	.082			−2000	−.0031	−3040	−1350	−3580
1701	12.4	.078	—	—	−1500	−.0026	−1520	−1140	−2400
CROWN									
1200	14.0	.094	5000	.0058	−1800	−.0021	−2310	−1460	−3210
1340	14.0	.094	3670	.0044	−1330	−.0016	−1610	−1100	−2430
1520	14.0	.078	2500	.0044	−900	−.0016	−1180	−680	−1990
1701	12.4	.078	1950	.0034	−900	−.0016	−1180	−680	−2410

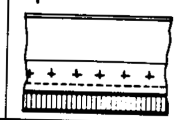
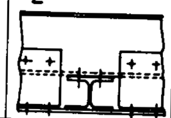
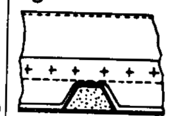
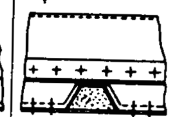
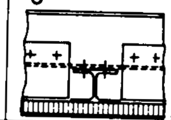
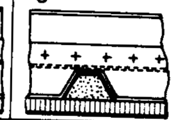
[1] ALLOWABLE TENSION STRAIN = 0.006 in/in

[2] ALLOWABLE COMPRESSION STRAIN = −0.005 in/in

[3] SMEARED THICKNESS DOES NOT INCLUDE HONEYCOMB MATERIAL

[4] TENSION LOADING IN KEEL DOES NOT INFLUENCE DESIGN

Figure 3.1-5. Analysis of Hat Section Stiffened Honeycomb Skin Design (Concept 6)

	ALUMINUM BASELINE WEIGHT, lb	COMPOSITE GR-EP CONCEPT WEIGHT, lb					
		1 	2 	3 	4 	5 	6 
SKIN PANELS	1710	1870	1560	1370	1370	1620	1620
STRINGERS	860	0	390	525	525	290	320
FRAMES	700	550	530	435	545	490	420
SKIN SPLICE PLATES	40	70	0	0	0	30	60
STRINGER SPLICE FITTINGS	50	0	0	0	0	0	0
WINDOW FRAMES	140	110	110	110	110	110	110
STA 1180 SPLICE MATERIAL	30	20	20	20	20	20	20
SEAM SEALANT	70	20	20	20	20	20	20
CORROSION INHIBITOR	10	0	0	0	0	0	0
POST ASSEMBLY PROTECTIVE FINISH	40	0	0	0	0	0	0
TOTAL WEIGHT*	3650	2640	2630	2480	2590	2580	2570
% WEIGHT REDUCTION	BASELINE	28	28	32	29	29	30

* FOR STUDY SECTION 540 INCHES LONG

Figure 3.2-1. Weight Study

GRAPHITE-EXPOXY COMPOSITE SHELL CONCEPT	BASIC FACTORY LABOR NORMALIZED HOURS [1]
CONCEPT 1 HONEYCOMB SANDWICH SKIN NO STRINGERS	1000
CONCEPT 2 LAMINATE SKIN I-SECTION STRINGERS	1050
CONCEPT 3 [2] LAMINATE SKIN HAT SECTION STRINGERS	1040
CONCEPT 5 HONEYCOMB SANDWICH SKIN I-SECTION STRINGERS	1280
CONCEPT 6 HONEYCOMB SANDWICH SKIN HAT SECTION STRINGERS	1400

[1] RELATIVE LABOR HOURS BASED ON FABRICATION OF CONSTANT SECTION WITH BODY FRAMES AT 20-INCH SPACING

[2] CONCEPT 3 and 4 SIMILAR

Figure 3.3-1. Labor Requirements for Composite Fuselage Fabrication

3.4 DESIGN SELECTION

Two concepts have been selected that merit further consideration for composite fuselage applications. These concepts are the full-depth honeycomb design with no stringers, Concept 1; and the I-section stiffened laminate skin design, Concept 2. These concepts represent two fundamentally different approaches to fuselage design in that the honeycomb concept is designed buckling resistant to the design ultimate load (DUL), while the skin in the stiffened laminate is designed to buckle at 30% DUL.

The foam filled hat section designs, Concepts 3 and 4, were not selected even though the relative weights and costs were better than the I-section stringer. An extensive inspection evaluation was performed on the foam filled hat section stringer and the results showed that the foam filled hat stringer panels could not be adequately inspected by current technology. Ultrasonic through transmission sound waves that are used during inspection are attenuated through the foam material, thus obscuring any detection signals. Other inspection methods, such as ultrasonic pulse echo, radiography, thermal imaging, and optical laser holography, do not provide adequate inspection quality for the foam filled stringers at this time. Concepts 5 and 6 were not considered due to the high cost even though they are weight competitive.

The inspection concerns described above are applicable to the foam filled window frame concepts shown in Figure 2.6-12. The solid laminate window frame designs shown in Figure 2.6-13 are fully inspectable, and therefore merit further consideration.

3.5 TOTAL FUSELAGE WEIGHT REDUCTION

The weight reduction for a total graphite composite aircraft fuselage has been estimated, based on the percent weight reductions established for the I-section stiffened (Concept 2) laminate skin design. The results are summarized in Figure 3.5-1. The weight reduction values for the composite design of the study section, shown in Figure 3.2-1 for Concept 2, were extrapolated to a full length aluminum fuselage structure on a component by component basis. Weight reductions were applied to fuselage components that could potentially be made with composites.

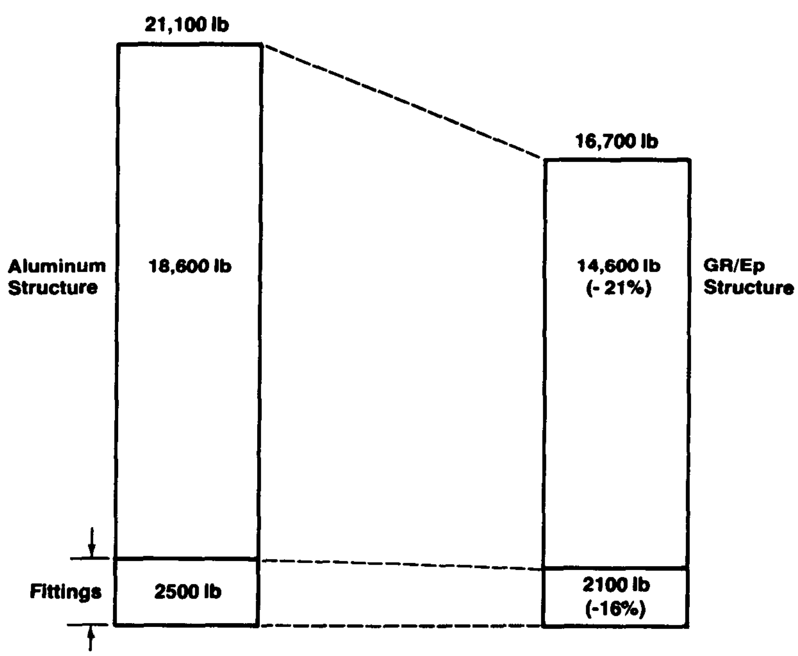


Figure 3.5-1. Commercial Fuselage Weight Reduction

The aluminum baseline fuselage structure weights used in this exercise are typical of an advanced technology, standard body, single aisle, medium range Boeing commercial aircraft. In this airplane fuselage weight study, 71% of the total structure weight was considered to be candidate structure for the use of graphite-epoxy material. The components not considered as candidate structure included windshields, windows, seat tracks, and components that are presently fabricated from composite materials such as wing-to-body fairing. The overall weight reduction applicable to the candidate structure was 21%, or 4000 pounds. Aluminum fittings totaling 2,500 pounds were separately identified. In redesigning load paths for a composite fuselage, an estimated 16% of the fitting weight could be removed.

To enable a realistic weight reduction forecast to be made, a comprehensive, Boeing production and IR&D aluminum fuselage structural component weight tabulation was utilized. Each component of the baseline structure was assessed and where a percentage weight reduction was judged not feasible for a direct application, the component was broken down into subcomponents and details. For example, for Concept 2, a 24% weight reduction resulted for fuselage frames and this reduction was applied to all standard frames. However, in the fuselage, special frames such as major support bulkheads include a significant weight of fittings. In these cases, the weight of the fittings associated with these frames was subtracted from the component weight, on the assumption that these fittings would either remain unchanged or would be replaced with fittings or structure of a similar weight. The 24% weight reduction was then applied to the remaining frame structure and the fitting weights were then added back to the reduced frame weight.

This approach was continued throughout the total fuselage structure with some 500 components being involved. Particular attention was given to entry and cargo doors and to bulkheads, as these components were not considered in the study section. For instance, a passenger door installation from the aluminum baseline fuselage was broken down to 56 detail parts and appropriate weight reductions were made where possible. However, 35 of these detail parts were either fittings such as hinges, stops, latches, snubbers, and so forth, or details that would remain unchanged and not included. The resulting overall weight reduction to the door, including the door surround structure, was 8%. This value was applied to all passenger, galley, cargo, and access doors. Passenger floor panels and floor support structure were also included as candidate structure in this extrapolation. The weight reductions used were based on previous Boeing IR&D study results.

4.0 MILITARY BENEFITS

Benefits from the application of graphite-epoxy composites and aluminum-lithium alloy to the fuselage structure of a military transport aircraft were determined.

4.1 BASELINE AIRCRAFT

A medium range tactical transport was selected as the baseline military aircraft for comparative studies. A drawing of the aircraft and its specifications is shown in Figure 4.1-1 and a side view that defines the fuselage body section are shown in Figure 4.1-2. The weight distribution of the major structural components in the aircraft and the total structural weight are shown in Figure 4.1-3. The design loads for the military aircraft fuselage, which were used for comparative sizing with the commercial aircraft baseline, are shown in Figure 4.1-4.

4.2 FUSELAGE WEIGHT REDUCTION

The weight reductions for the medium range military tactical transport fuselage were calculated by extrapolating the weight reductions established for the commercial baseline. The design loads for the military transport (fig. 4.1-4) and the commercial transport (fig. 2.5-2) are in the same range. Therefore, the skin stringer and frame weight savings for the military transport would be similar to the commercial transport as shown in Figure 3.2-1. The Concept 2 I-section stringer design was used for the extrapolation. The detailed procedure used for the extrapolation is identical to that described for the commercial aircraft fuselage (see sec. 3.5). For the military aircraft, the cargo floor, walkway, and ramp floor were not considered candidate structure for composites due to the highly localized service loading.

The weight of the total fuselage is 55,640 pounds of which 35,400 pounds was considered as candidate structure. The extrapolation procedure produced a 19% reduction of 6900 pounds as shown in Figure 4.2-1. An additional 600-pound reduction in fitting weight was identified in a manner similar to that described for the commercial transport.

A comparable analysis of weight reduction was performed considering the fuselage fabricated from aluminum-lithium alloy. Of the 55,640 pounds of fuselage weight, candidate structure totaling 44,670 pounds was identified. The nonparticipating structure included windows, windshields, existing composite structure and existing nonaluminum parts in the cargo floor, cargo floor support structure, and loading ramp. Assuming an 8% change due to the lower density, a reduction of 3600 pounds would be realized.

4.3 FLEET SERVICE BENEFITS

The potential benefits that would be realized for a fleet of tactical military aircraft from the application of graphite composites or aluminum-lithium to the fuselage structure were determined. The potential benefits were estimated by three different methods based on a differing set of assumptions.

For the first method, calculations based on a constant fleet size were used to determine how the structural weight reduction would reduce fleet fuel consumption. The detailed assumptions are defined as follows:

- Baseline and advanced military fleets contain 200 airplanes each (assumed size of peacetime tactical transport fleet)
- Payload capability is constant

- Weight reduction is reflected directly as a gross weight reduction that results in direct fuel savings
- Support and maintenance costs for the advanced fuselage military fleet are the same as the baseline fleet
- Total cost savings, resulting from fuel savings, is based on using typical peacetime flight hours per year per airplane of 1168 hours
- Direct fuel savings per 1000 pounds of weight reduction is 60 pounds per hour
- Fuel cost is $1.176 per gallon
- Weight of fuel is 6.5 pounds per gallon
- Service life is 20 years

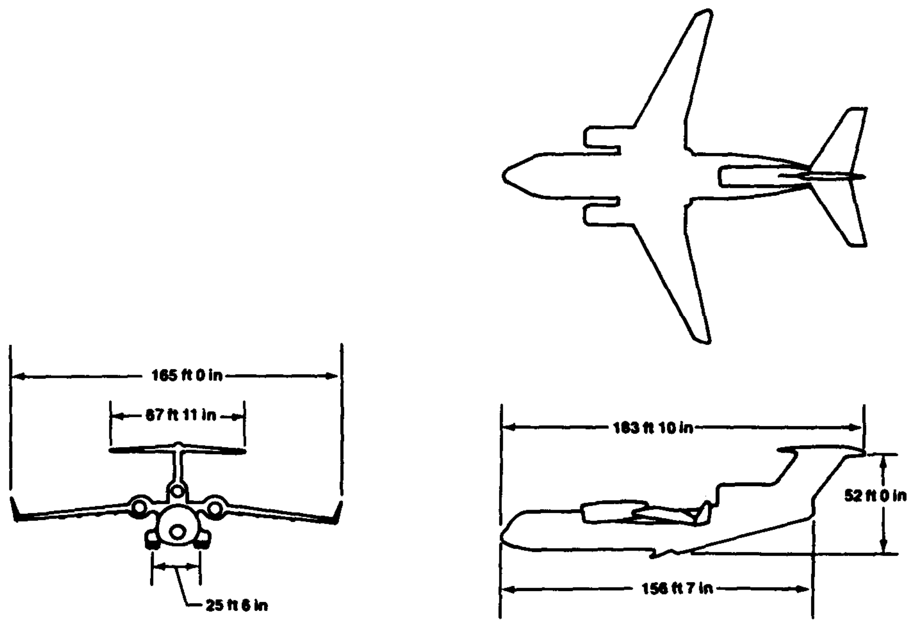
The results of this analysis are shown in Figure 4.2-2.

The second method used to define benefits assumes that the fleet size would be reduced, while maintaining total fleet payload lifetime capacity constant, since the reduction in structural weight would translate directly into an increase in payload per airplane. The detailed assumptions used in this method are defined as follows:

- Weight reduction is translated directly into an equal amount of payload increase while keeping the gross weight unchanged
- The increased payload capability is fully utilized by all airplanes of the fleet
- The payload capacity per baseline airplane is 140,000 pounds
- Operation and support costs per airplane is the same for the baseline and advanced fuselage airplane
- Any reduction in fleet size results in corresponding operation and support cost savings
- Operation and support costs do not vary with acquisition costs
- Acquisition cost per airplane is the same for the baseline and the advanced fuselage airplane
- 200 airplanes are in the baseline fleet
- Estimated value for operation and support costs per airplane per service life is $\$104 \times 10^6$

Total life cycle cost savings for a fleet of military transport airplanes is determined based on an acquisition cost reduction combined with an operation and support cost reduction due to a fleet reduction. The life cycle cost reductions, shown in Figure 4.2-3, are calculated for three assumed values of acquisition cost.

The third method used to define benefits assumes that the payload remains fixed and the takeoff gross weight (TOGW) is reduced, which results in improved performance. Performance factors for the military transport were determined for the weight reduction and the changes in the transport performance are shown in Figure 4.2-4. The fuel consumption rate would be reduced, which would extend the range. Due to the lower TOGW, the normal field length and the austere mission field length would be reduced as shown.



PRINCIPAL CHARACTERISTICS	
MAXIMUM TAXI WEIGHT (BASIC)	490,000 lb
MAXIMUM TAKEOFF WEIGHT	490,000 lb
MAXIMUM LANDING WEIGHT	397,700 lb
MAXIMUM ZERO FUEL WEIGHT	326,200 lb
ENGINE THRUST	48,000 lb
FUEL CAPACITY	30,200 lb
CARGO CAPACITY	
ALL BULK	16,144 ft^3
MAXIMUM OPERATING SPEED	320 knots
MACH NUMBER	0.80

Figure 4.1-1. Military Baseline Model

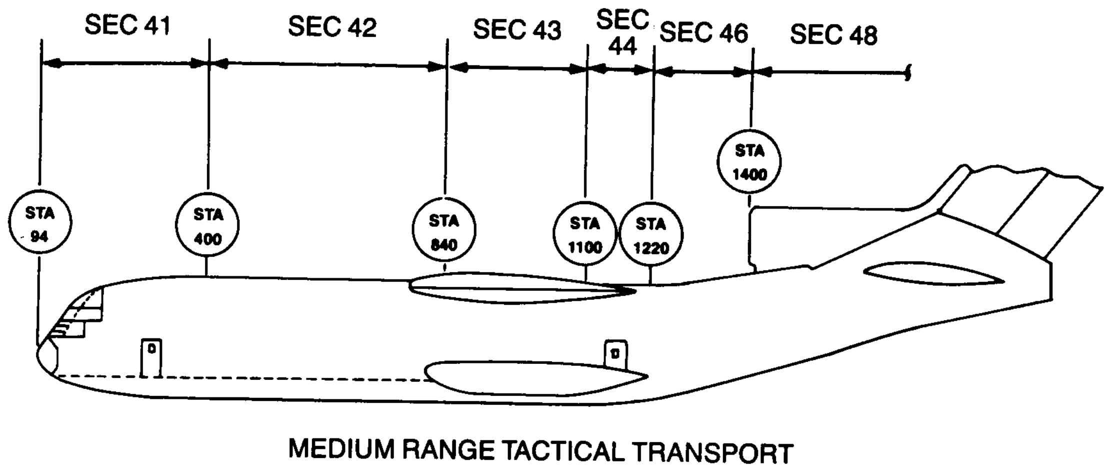


Figure 4.1-2. Military Transport Baseline Fuselage

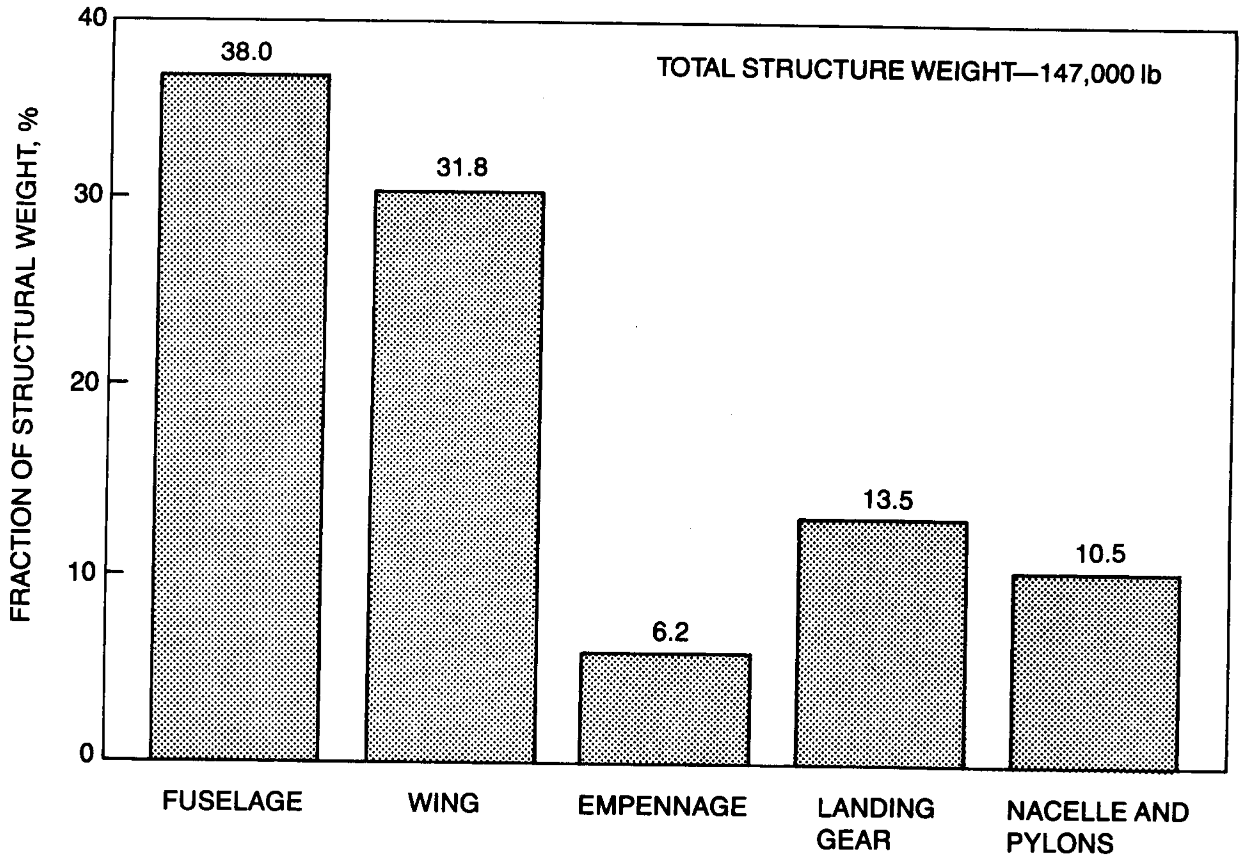


Figure 4.1-3. Military Baseline Component Weight Distribution

BODY STATION	NEGATIVE BENDING [1] CROWN IN TENSION		POSITIVE BENDING [1] CROWN IN COMPRESSION		SIDE PANEL SHEAR FLOW, q LB/IN
	CROWN PANEL LOAD, Nx LB/IN	KEEL PANEL LOAD, Nx LB/IN	CROWN PANEL LOAD, Nx LB/IN	KEEL PANEL LOAD, Nx LB/IN	
400	510	– 445	–1065	940	740
500	1165	–1025	–1620	1430	865
600	1925	–1695	–2030	1780	1090
700	2890	–2540	–2330	2050	1185
800	3750	–3300	–2485	2190	1310
840	4155	–3655	–2530	2230	1490
1100	6335	–5575	–1875	1650	2050
1220	5065	–4460	–2130	1870	1650

[1] SEE FIGURE 2.5-1 FOR SIGN CONVENTION.

Figure 4.1-4 Fuselage Design Loads for Medium Range Tactical Transport

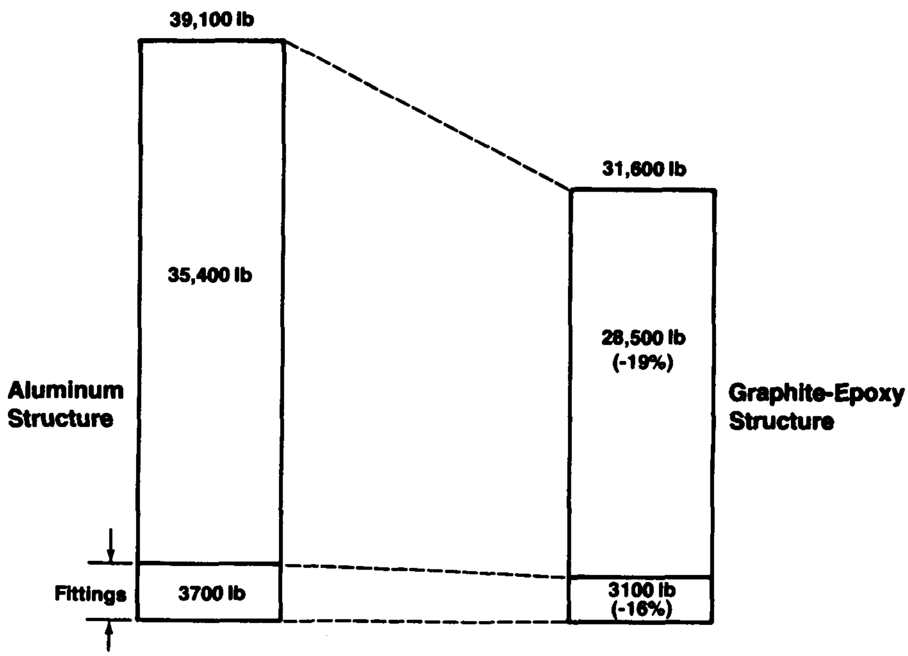


Figure 4.2-1. Military Fuselage Weight Reduction

ADVANCED AIRPLANE	WEIGHT REDUCTION, lb	FUEL SAVED PER AIRPLANE, 10^6 lb	FLEET FUEL SAVINGS, 10^6 lb	VALUE OF FLEET FUEL SAVING, 10^6 DOLLARS
GRAPHITE COMPOSITE FUSELAGE	7500	10.5	2100	380
ALUMINUM LITHIUM FUSELAGE	3600	5.0	1010	180

COST SAVINGS FOR FLEET OF 206 AIRPLANES FOR 20 YEAR SERVICE LIFE
(CONSTANT YEAR DOLLARS)

Figure 4.2-2. Tactical Transport Fleet Fuel Savings

ADVANCED AIRPLANE	REDUCTION IN FLEET SIZE	FLEET COST REDUCTION 10^6 DOLLARS		
		ACQUISITION COST $\$30x10^6$	ACQUISITION COST $\$40x10^6$	ACQUISITION COST $\$50x10^6$
GRAPHITE COMPOSITE FUSELAGE	10	1340	1440	1540
ALUMINUM LITHIUM FUSELAGE	5	670	720	770

COST SAVING BASED ON ASSUMED ACQUISITION COST AND OPERATION AND SUPPORT COST OF $\$104x10^6$ PER AIRPLANE PER SERVICE LIFE (CONSTANT YEAR DOLLARS)

Figure 4.2-3. Tactical Transport Reduced Fleet Size Cost Saving

	BASELINE ALUMINUM	COMPOSITE FUSELAGE	COMPOSITE FUSELAGE REDUCTION	ALUMINUM LITHIUM FUSELAGE	ALUMINUM LITHIUM FUSELAGE REDUCTION
OPERATING WEIGHT EMPTY, lb	221,200	213,700	7500	217,600	3600
TAKEOFF GROSS WEIGHT, lb	490,000	482,500	7500	486,400	3600
FUEL FLOW, lb/hr	15,280	14,871	409	15,084	196
FERRY RANGE, nmi	6,010	6,215	−205	6,108	−98
NORMAL TAKEOFF DISTANCE, ft	7,600	7,285	315	7,449	151
AUSTERE FIELD TAKEOFF DISTANCE, ft	2,800	2,643	157	2,725	75

Figure 4.2-4. Tactical Transport Fleet Increased Performance

5.0 MANUFACTURING DEVELOPMENTS

5.1 MANUFACTURING METHODS

The concepts defined in Section 2.6 were evaluated to assess their manufacturing risk and technology developments required to minimize manufacturing costs. Fabrication of the detail parts was the primary factor used to evaluate the concepts. The fabrication assessment included the tooling approach required by the concept and the availability of automated fabrication methods.

A typical manufacturing flow, shown in Figure 5.1-1, includes processes for laying up, trimming, curing, inspecting, and assembling parts. The manufacturing sequence planned for laminate stiffened panels is summarized in Figure 5.1-2 and the manufacturing sequence planned for honeycomb panels is summarized in Figure 5.1-3. The sequence planned for the stringer stiffened honeycomb panels would be a combination of the sequences shown in Figures 5.1-2 and 5.1-3. Procedures for fabrication, assembly, and inspection that would be used in these manufacturing flows are discussed below.

5.1.1 Fabrication

For the laminate skin concepts, flat tape laminating by automation and numerically controlled (NC) trimming would be used. These methods are currently used as shown in Figures 5.1-4 and 5.1-5. Parts that have been made by these methods, however, are relatively small in area compared to a full-scale fuselage skin. If the fuselage skins are laid up flat and transferred to the final curing tool, then transfer techniques will have to be developed. Automated methods to lay the tape material directly into the final curved shape have been considered, but the equipment necessary to perform this task is not available and would have to be developed. The tear strap details, discussed in Section 2.8, were considered to be laid down by the flat tape laminator as an integral step in the skin buildup.

Filament winding the laminate skin material on a mandrel and then slitting and transferring the material to the final cure tool is a method that has been considered but has not been developed for fuselage size parts. Filament winding and curing the laminate skin material on a mandrel has been considered as an automation method, but, again, this procedure needs to be developed and verified.

The manufacturing method considered for fabrication of honeycomb panels was to laminate both the inner and outer skins on the flat tape laminator and transfer the laminates to the final cure tool. The fabrication sequence is defined in Figure 5.1-3.

The method considered for fabricating I-stringers was to use a flat tape laminating machine for building up the laminate, and then NC trimming. The cut laminate is then draped over the stringer tool and the tool halves are assembled on the skin, and the entire assembly is then bagged and cured as shown in Figure 5.1-6. A photograph of a cocured I-section stringer panel is shown in Figure 5.1-7.

The hat stringer laminate is laid up over a foam core, which remains an integral part of the structure after curing (fig. 5.1-8). A photograph of a hat section stringer panel is shown in Figure 5.1-9.

The method considered for fabrication of the frames was to cut flat pattern sections from woven broad goods, by NC, drape into the tool, and bag and cure. A photograph of fuselage frames fabricated by this procedure is shown in Figure 5.1-10. Other methods of fabrication, including filament winding and resin transfer molding in matched metal dies, were considered, but these procedures have not been developed.

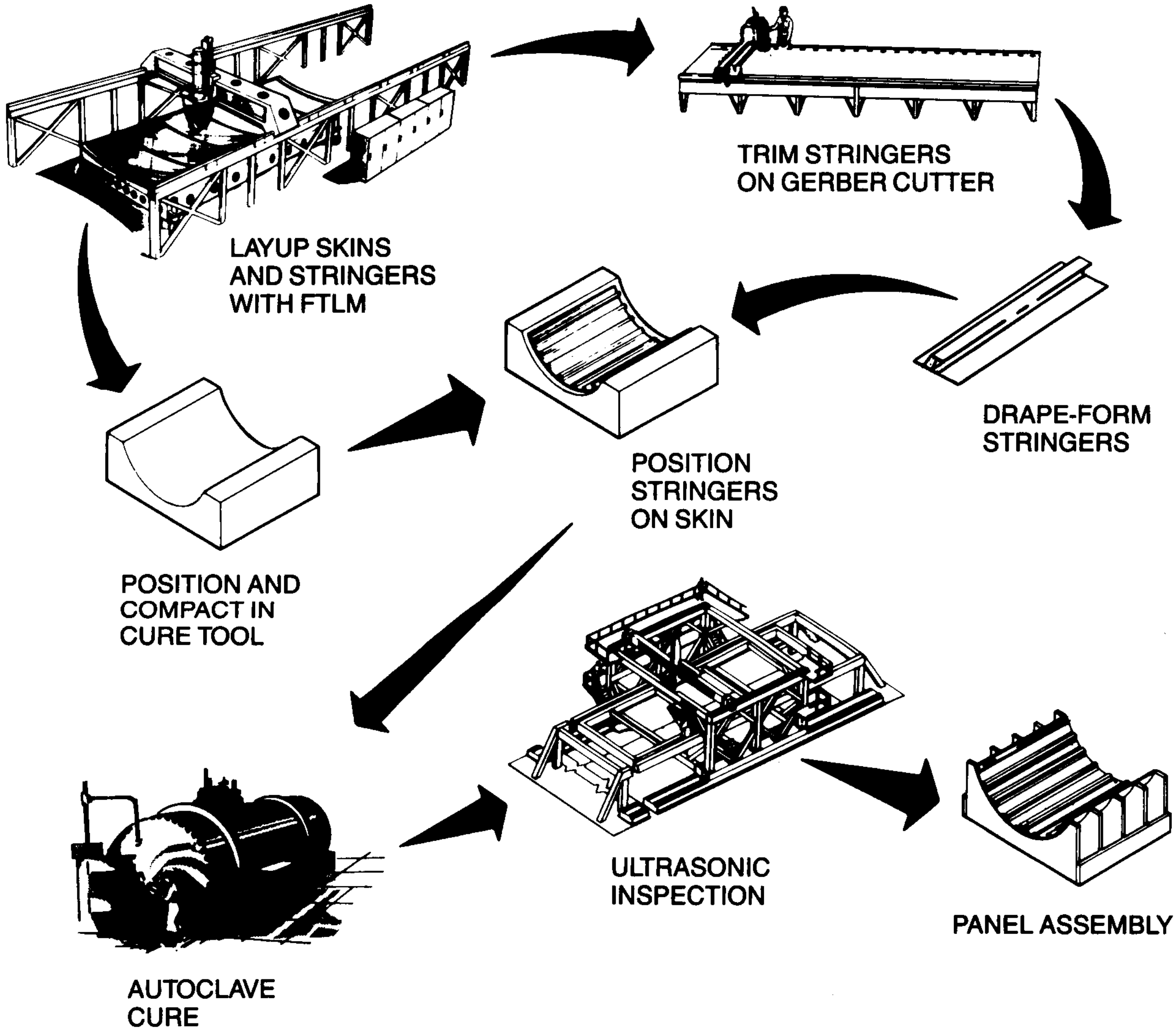


Figure 5.1-1. Typical Composite Fuselage Panel Manufacturing Flow

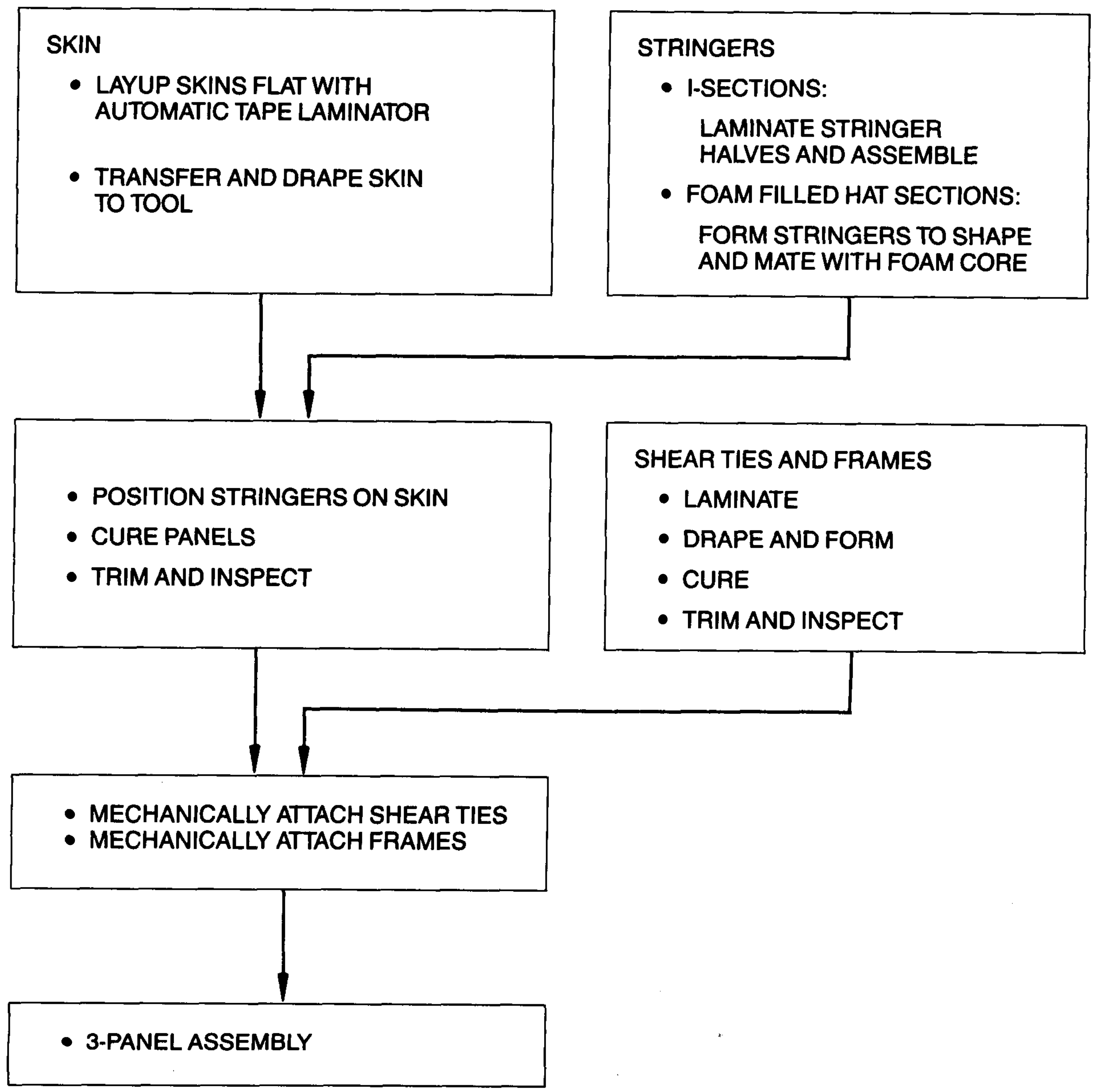


Figure 5.1-2. Fuselage Manufacturing Sequence for Laminate Stiffened Designs

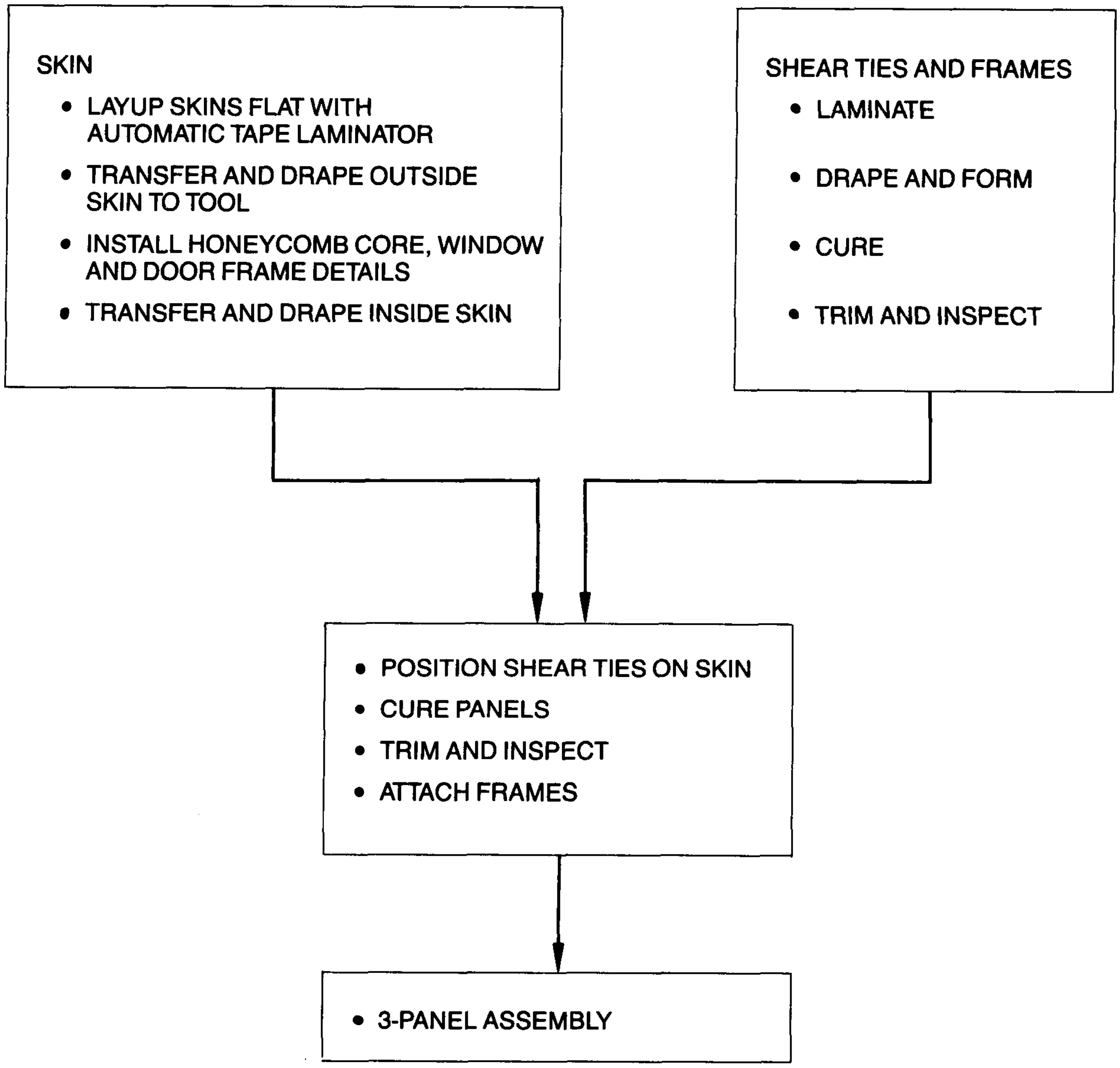


Figure 5.1-3. Fuselage Manufacturing Sequence for Honeycomb Skin Without Stringers

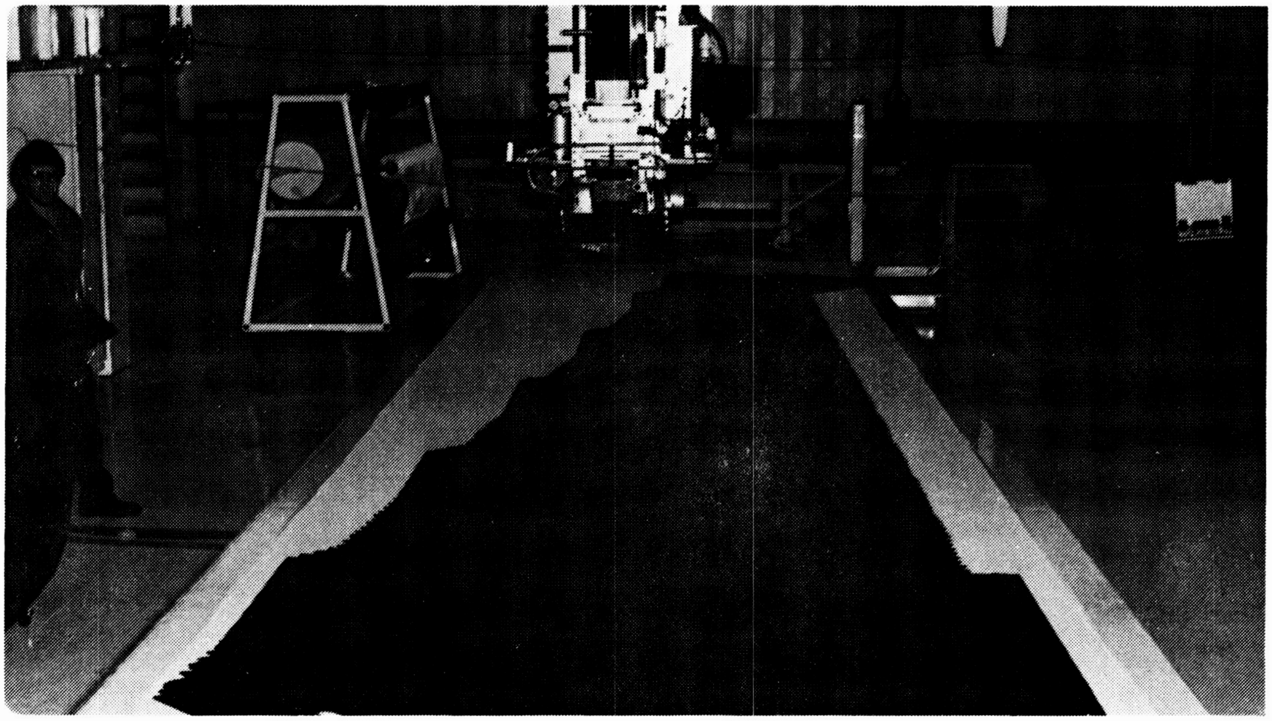


Figure 5.1-4. Automated Flat Tape Laminating Machine

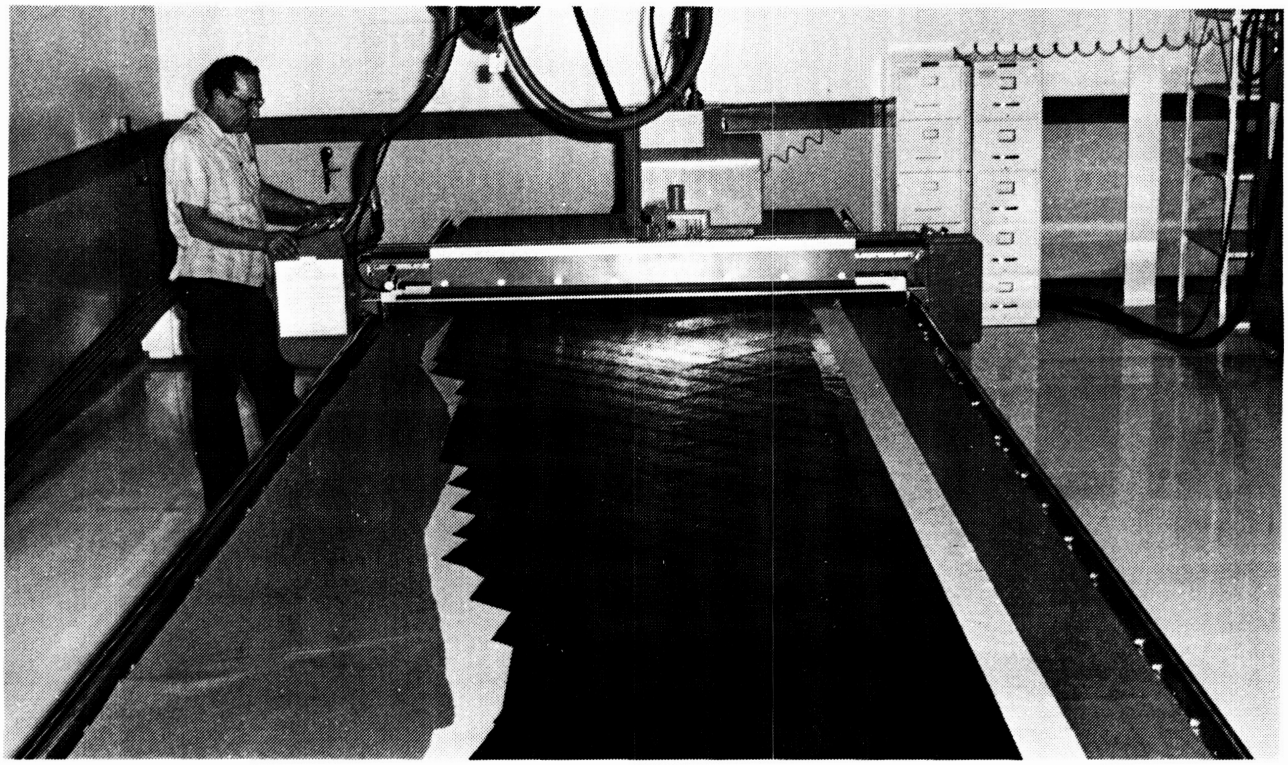


Figure 5.1-5. Numerically Controlled Cutter

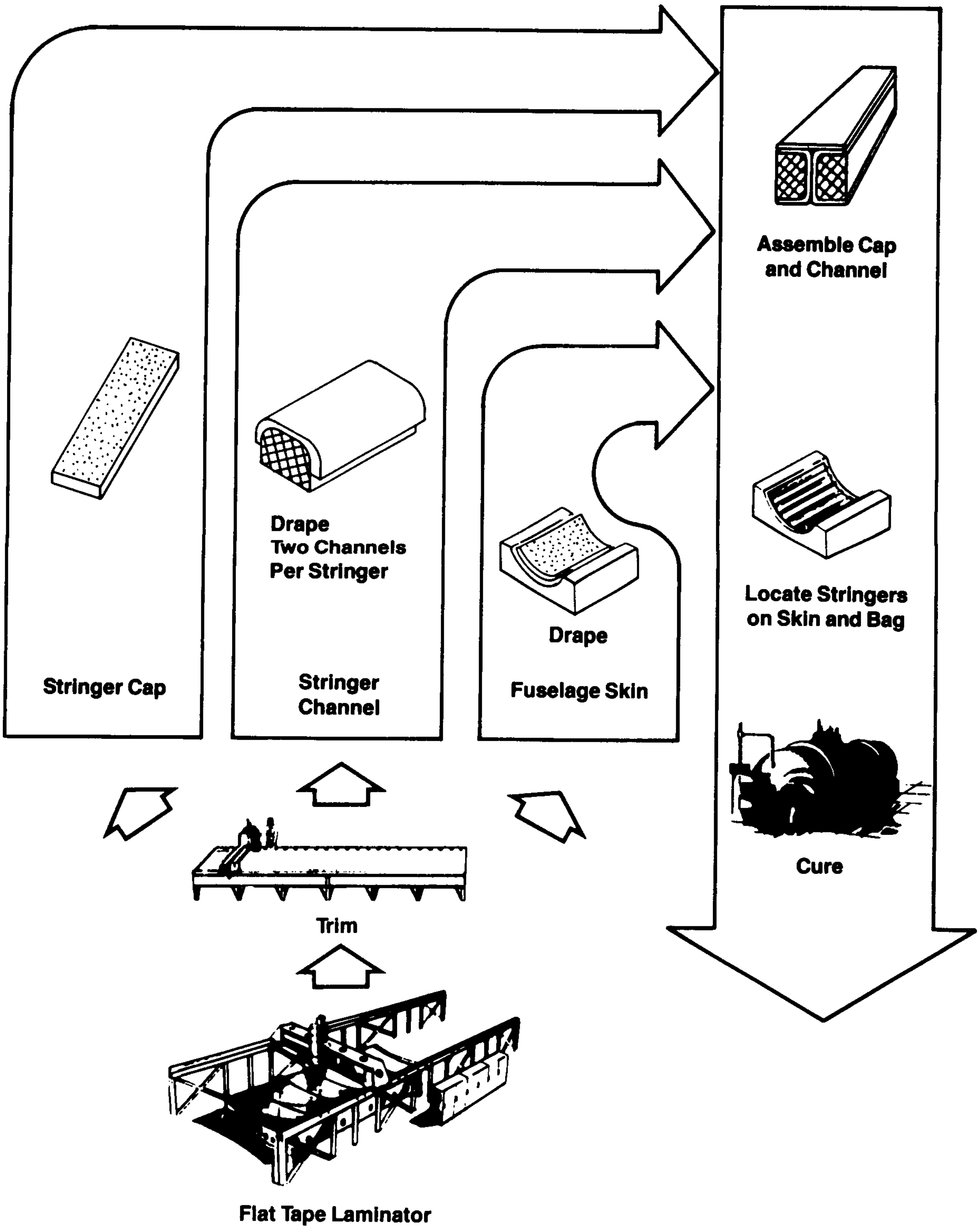


Figure 5.1-6. I-Section Stringer Panel Fabrication

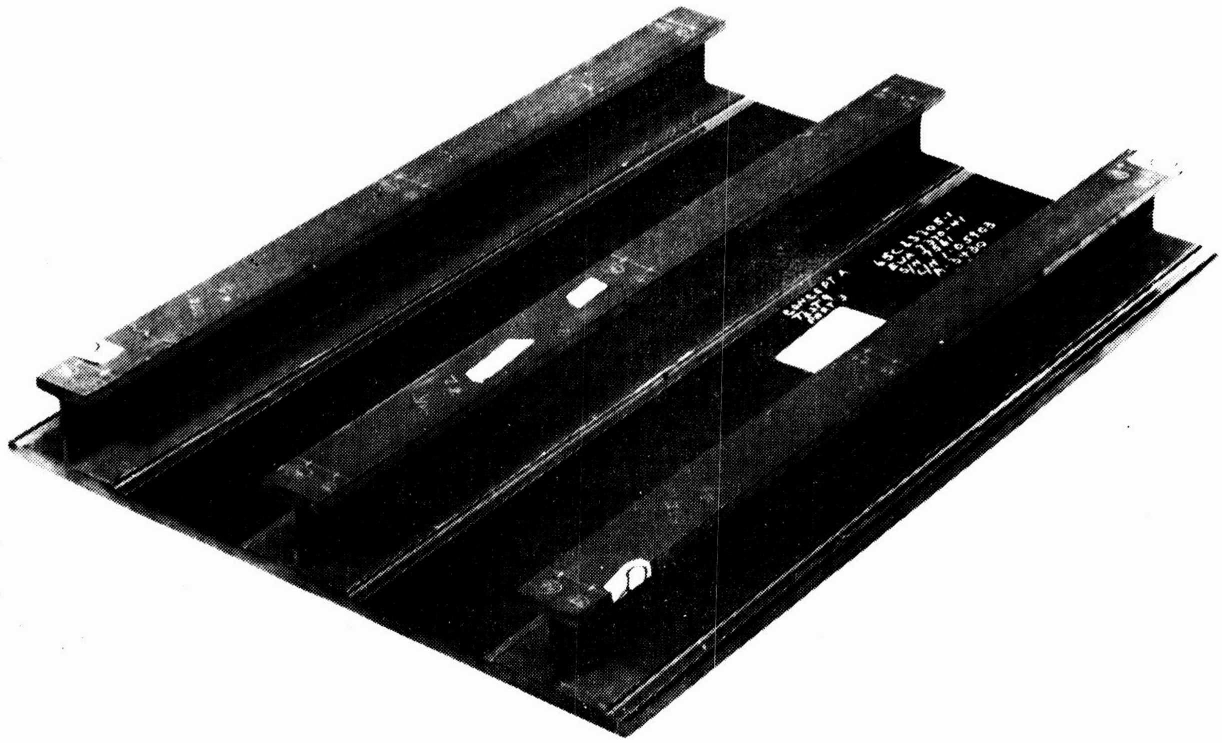


Figure 5.1-7. Cocured Graphite-Epoxy I-Section Stiffened Panel

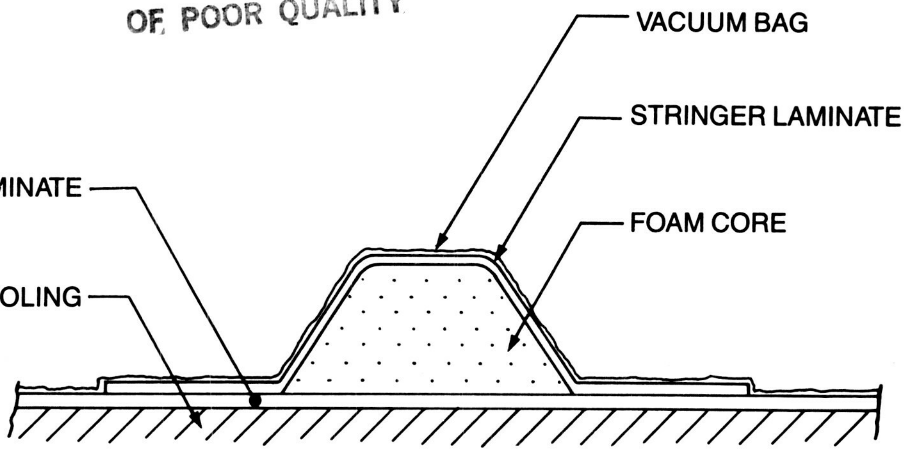


Figure 5.1-8. Foam Filled Hat Section Stringer Fabrication

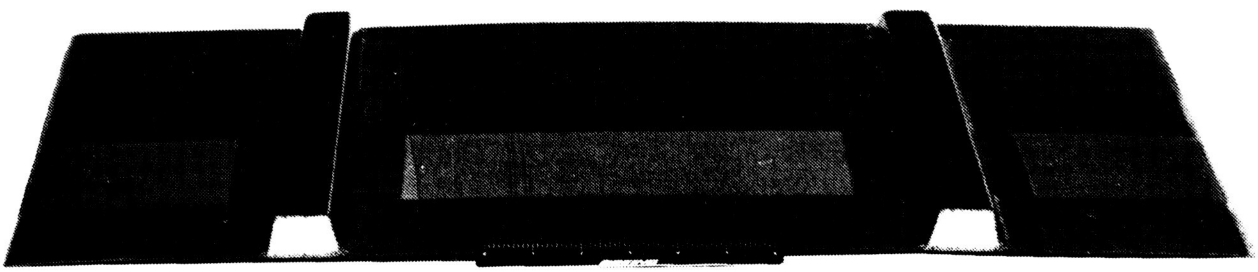


Figure 5.1-9. Graphite-Epoxy Foam Filled Hat Section

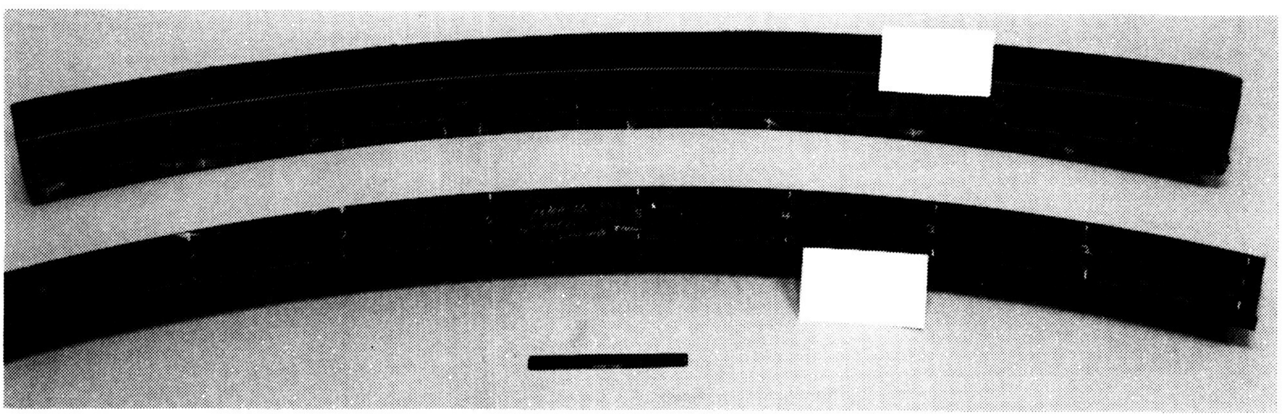


Figure 5.1-10. Graphite-Epoxy Body Frames

5.1.2 Assembly

Many of the parts that are generally fabricated and assembled separately in aluminum fuselage structures will be cocured in a composite fuselage. The skin of an aluminum fuselage shell structure, for example, requires numerous subassemblies, including skin panels, stringers, attachment clips, and so forth. The corresponding components in a composite fuselage can be cocured together during fabrication, which reduces part count. It is estimated that by cocuring, the part count in a composite fuselage shell can be reduced by as much as 20% of that of an aluminum fuselage shell.

The assembly sequence for composite fuselage structures is based on a three-panel barrel design. The three-panel design has the advantages of minimal longitudinal joints, while still keeping the panel size manageable. The assembly approach utilizes internal assembly tooling, shown in Figure 5.1-11. During the final stages of the fuselage panel assembly sequence, the keel and two side panels are set into assembly jigs and the jig segments are rotated up into the final position as shown in Figure 5.1-12. To reduce time and costs, the drilling for the panel longitudinal splice fasteners would be performed by an automated track drill, schematically shown in Figure 5.1-13.

5.1.3 Inspection

The quality assurance plan was to inspect all composite parts using state-of-the-art techniques of through transmission ultrasonics (TTU), pulse echo, and X-ray. An example of an automated TTU scanner is shown in Figure 5.1-14. In the critical area of the stiffener radius, automated scanning transducers, as shown in Figure 5.1-15, would be used.

5.2 MANUFACTURING EVALUATION

The design concepts defined in Section 2.6 were evaluated based on complexity, part count, and ease of automation. Simplification of part configuration improves the potential for automating fabrication, but often at the expense of increasing part count. In addition to incurring higher direct manufacturing costs, a higher part count increases bookkeeping, handling, and storage costs.

The principal advantages and concerns for manufacturing the design concepts are summarized in Figure 5.2-1. The labor requirements, discussed in Section 3.3 and shown in Figure 3.3-1, provide an assessment of the relative fabrication and assembly costs for the design concepts. The labor requirements combined with the advantages and concerns discussed in the following sections provided the basis for the design selection discussed in Section 3.4.

5.2.1 Full-Depth Honeycomb Sandwich Skin

The monocoque honeycomb structure skin design, Concept 1, is the simplest and least labor intensive of the six concepts. The overall assembly costs are kept to a minimum due to the low part count, and the honeycomb skin can be inspected using state-of-the-art automated techniques. During fabrication, minimal tooling is required for the skin. Skin face sheets can be laid up separately by automatic methods and transferred to the tool. During the cure process, distortion of the core is eliminated by limiting autoclave cure pressures to 45 psi. Cocuring the frames with the honeycomb skin complicates the fabrication process and this procedure was not considered. Cured part tolerances must be accurately controlled, since the stiffness of the honeycomb sandwich reduces the capability of movement to align parts during assembly.

5.2.2 Laminate Skin With Stringers

The layup, trimming, and inspection processes for the designs with laminate skins and stringers, Concepts 2, 3, and 4, show good potential for automation. For all of these concepts, the skins and stringers are laid up by automated tape laying machines and then cocured. With any stringer configuration, care must be taken during fabrication to ensure that the stringer centerline remains straight along the length of the panel.

Concept 2, which has an I-section stringer, uses hard tooling to define stringer shape. In addition to initial manufacturing expenses, the I-stringer tools incur additional labor requirements for handling and positioning during layup, and for removal from the part after curing. An advantage to the hard tooling is that the I-stringer can be cocured to the skin at a high autoclave pressure of approximately 85 lb/in^2. With this cure pressure, laminate porosity is minimized. In Concepts 3 and 4, tooling requirements are minimized since a foam core material is used to define the hat section stringer shape. Autoclave pressures need to be limited to avoid compacting the foam core materials, thus increasing the potential for laminate porosity.

The I-section stringer design (Concept 2) can be inspected by state-of-the-art techniques. The foam-filled hat section designs (Concepts 3 and 4), though, cannot be inspected by state-of-the-art techniques as discussed in Section 3.4.

The method used to attach body frames to the outer shell influences the complexity of both fabrication and assembly. Since mechanically attached frames can be cured separately from the shell, fabrication is simpler than with cobonded frames. Mechanically attached frames, though, incur higher assembly costs. With Concept 2, the frames can be fabricated separately and then mechanically attached directly to the flanges of the I-section stringers. Since the hat section stringers of Concepts 3 and 4 do not have accessible attachment points, the frame is attached to the skin via stringer clips machined to provide clearance over the stringers. The frame attachment methods used for Concepts 3 and 4 differ, as described in Section 2.6.2. In Concept 3, the fabrication process is complicated by cocuring a T-section to the skin. In Concept 4, a channel frame is mechanically attached directly to the skin.

5.2.3 Honeycomb Skin With Stringers

Concepts 5 and 6, which employ stringers cocured to honeycomb sandwich skins, are the most complex and costly of the designs to fabricate, assemble, and inspect (see fig. 3.3-1), and offer no manufacturing advantages over Concepts 1, 2, and 3.

5.3 TECHNOLOGY READINESS

As previously discussed in Section 5.1, current manufacturing methods were assumed for all concepts to arrive at a comparative evaluation. Other manufacturing methods that appear to have potential for reducing costs but have not been developed were discussed. The intent of the Air Force Mantech Fuselage Program (ref. 5.3-1) is to develop the most cost effective methods for fuselage fabrication and assembly. Several different procedures will be used to fabricate laminate skins, stringers, and frames. The most cost effective method will be selected and further evaluated for suitability for production.

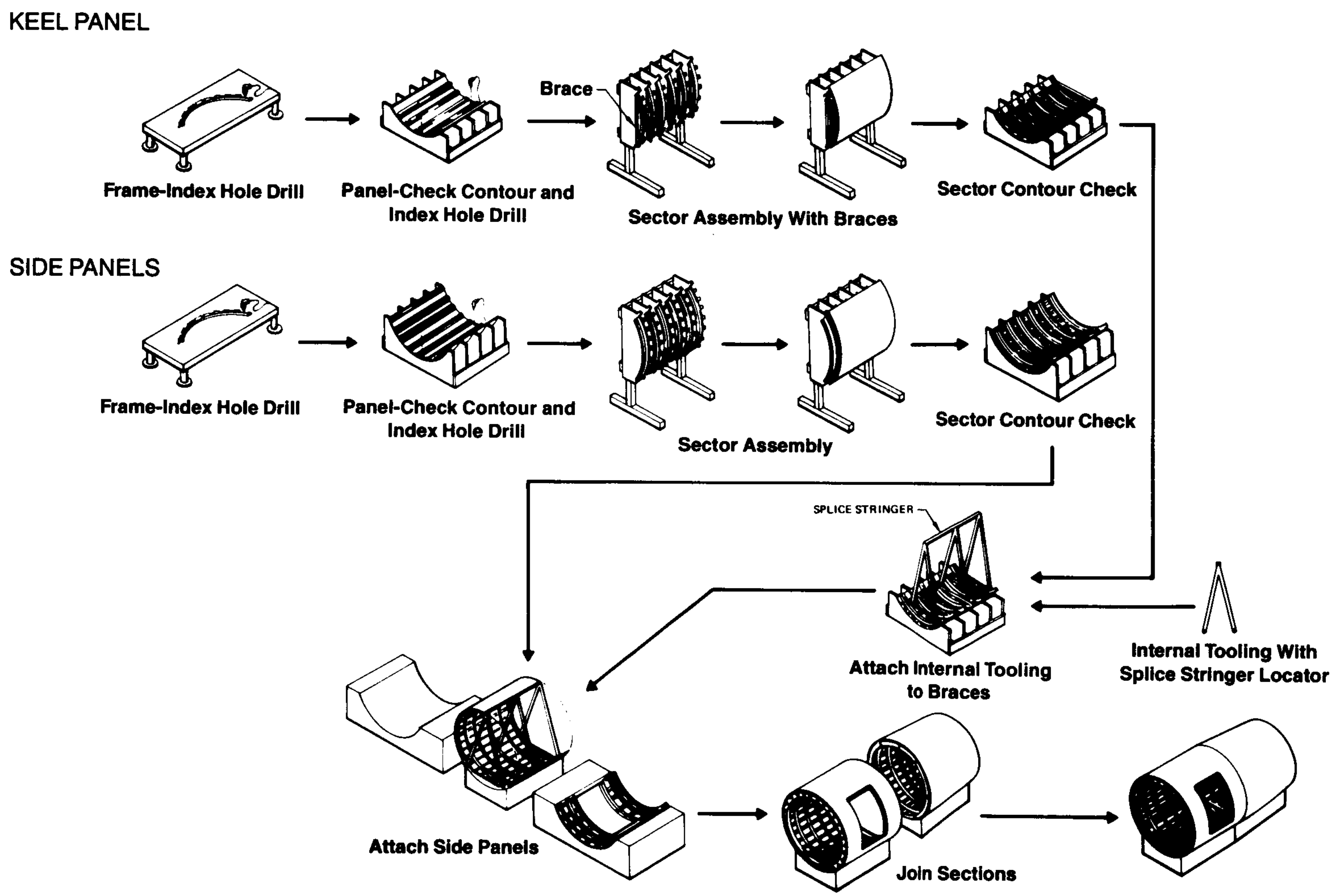


Figure 5.1-11. Fuselage Panel Assembly Process

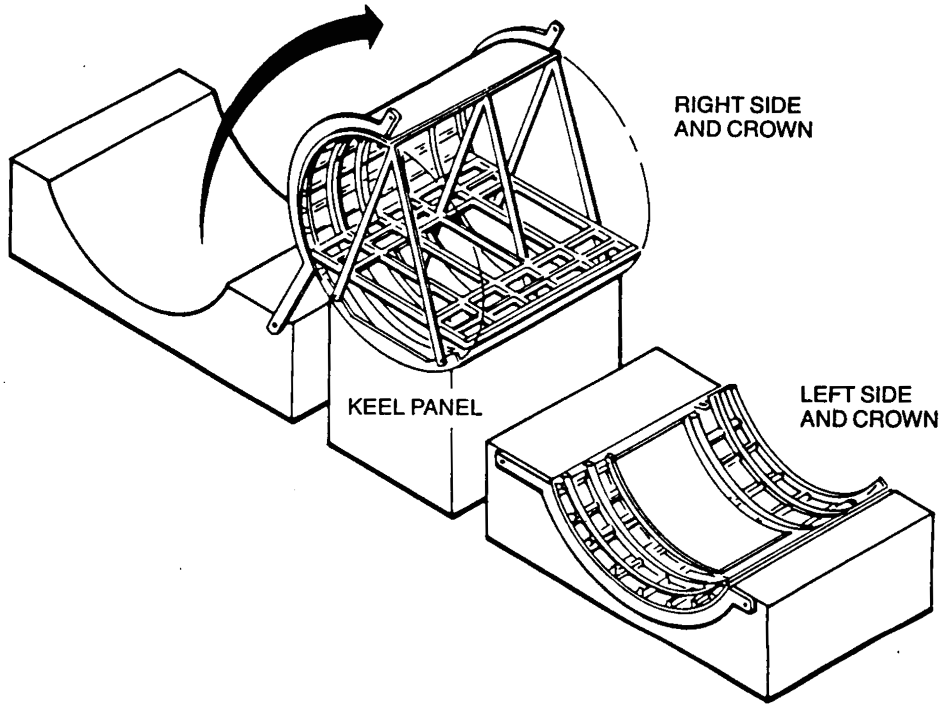


Figure 5.1-12. Automated Three-Panel Assembly

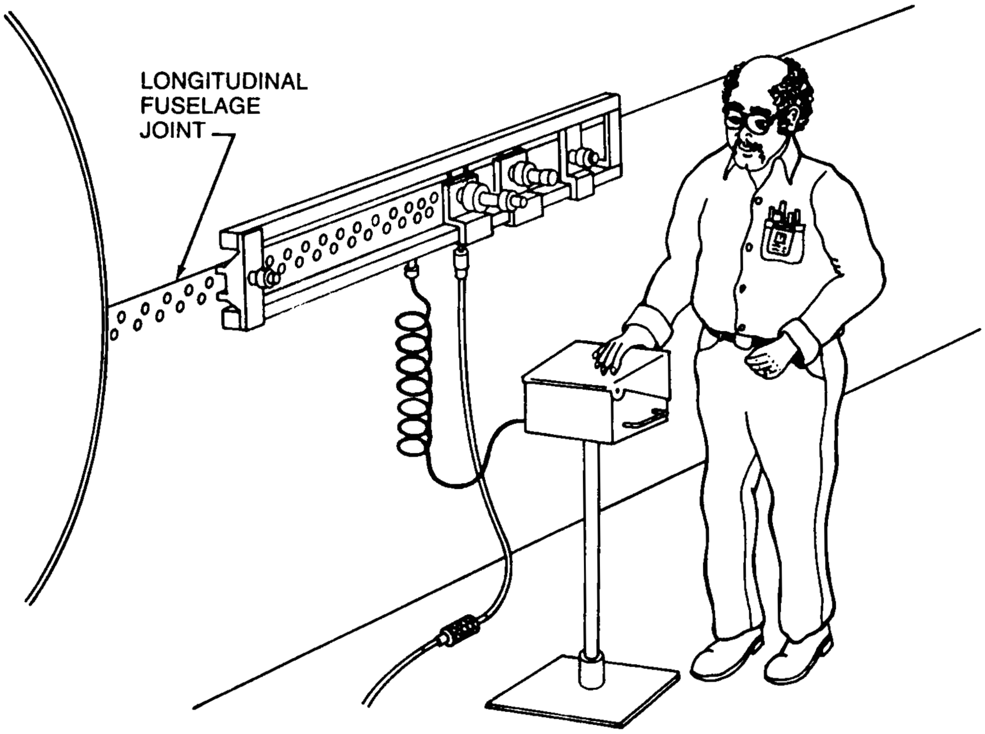


Figure 5.1-13. Automated Drilling Schematic

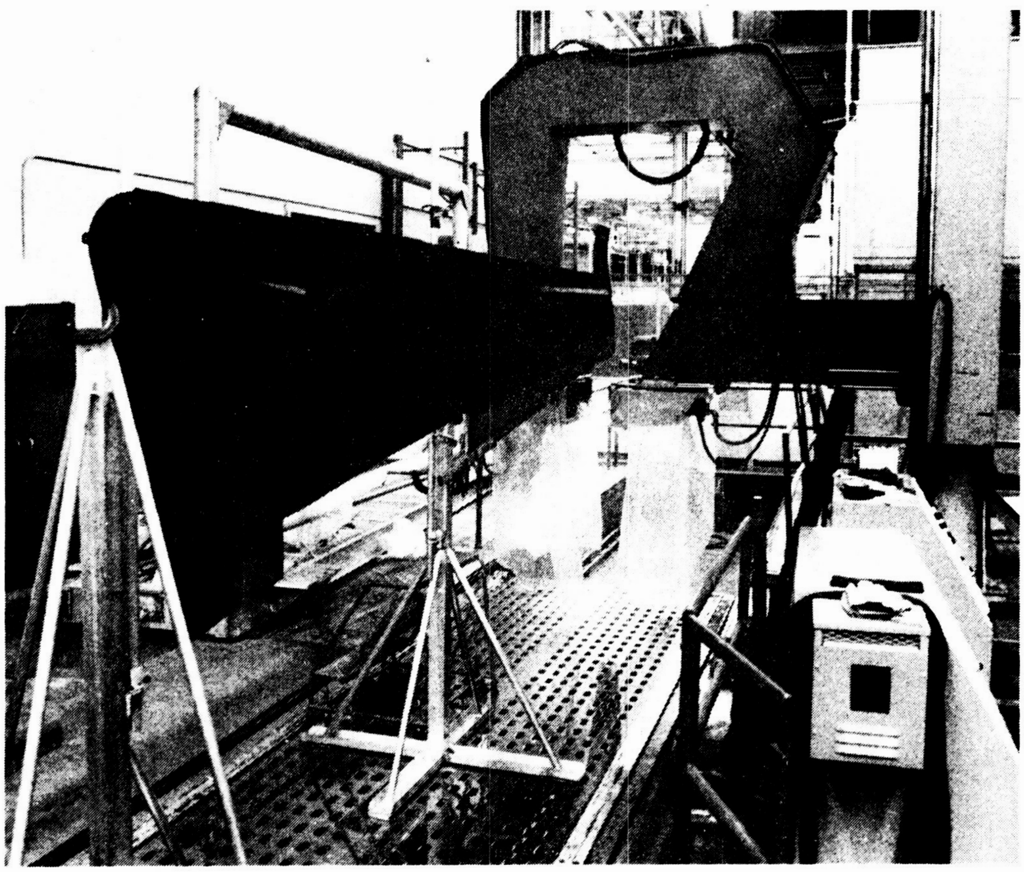


Figure 5.1-14. Through Transmission Ultrasonic Inspection

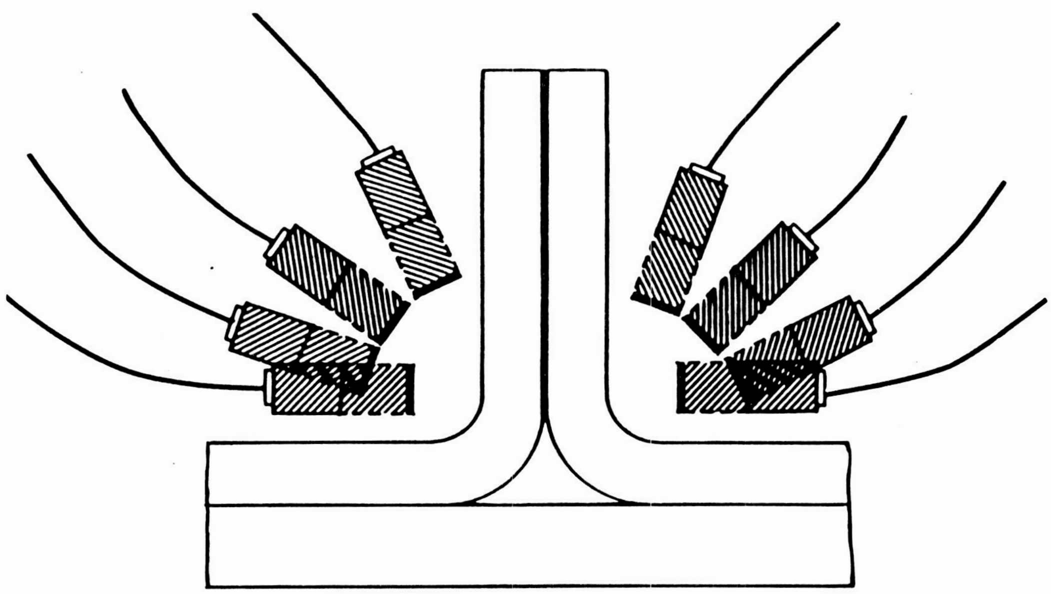


Figure 5.1-15. Transducer Array for NDE Inspection of Stringer Radius

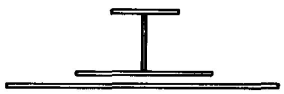
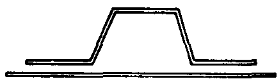
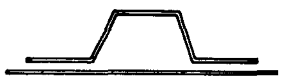
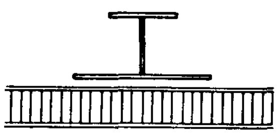
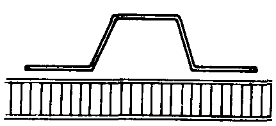
CONCEPT	ADVANTAGES	CONCERNS
1 	• MINIMAL TOOLING • LOW PART COUNT • INSPECTABLE	• MINIMAL ASSEMBLY TOLERANCE PAYOFF
2 	• AUTOMATABLE • HIGH PRESSURE CURE • INSPECTABLE • ACCESSIBLE FOR ASSEMBLY	• COMPLEX STRINGER TOOLING • STRINGER CENTERLINE CONTROL
3 BONDED FRAME 	• AUTOMATABLE • MINIMAL TOOLING REQUIRED	• AUTOCLAVE PRESSURE LIMITATIONS • INSPECTION • STRINGER CENTERLINE CONTROL • COMPLEX FRAME BONDING TOOLING
4 MECHANICALLY ATTACHED FRAME 	• AUTOMATABLE • MINIMAL TOOLING REQUIRED	• AUTOCLAVE PRESSURE LIMITATIONS • INSPECTION • FASTENING THROUGH HAT STRINGER • STRINGER CENTERLINE CONTROL
5 	• INSPECTABLE • ACCESSIBLE FOR ASSEMBLY	• LOW AUTOMATION POTENTIAL • COMPLEX TOOLING • MECHANICAL ATTACHMENTS • AUTOCLAVE PRESSURE LIMITATIONS • STRINGER CENTERLINE CONTROL
6 	• MINIMAL TOOLING	• LOW AUTOMATION POTENTIAL • INSPECTION • MECHANICAL ATTACHMENTS • AUTOCLAVE PRESSURE LIMITATIONS • STRINGER CENTERLINE CONTROL

Figure 5.2-1. Manufacturing Evaluation of Design Concepts

6.0 TECHNOLOGY ISSUES

The technology issues facing composites application to fuselage structure are separated into areas relating to materials, structures, systems, and manufacturing. These issues must be addressed simultaneously with the advanced composite fuselage design development.

6.1 MATERIALS

Material usage investigations to date have primarily addressed empennage and wing structures. Studies and evaluations of composite materials are needed for fuselage structure. Optimum composite material systems need to be identified for both solid laminate and sandwich structure. In addition, there are supplementary materials that will be required in fuselage designs. Such materials include:

- Honeycomb and other lightweight core materials for sandwich structure
- High strength potting compounds for attachment and reinforcement in honeycomb structure

6.1.1 Flammability and Fire Protection

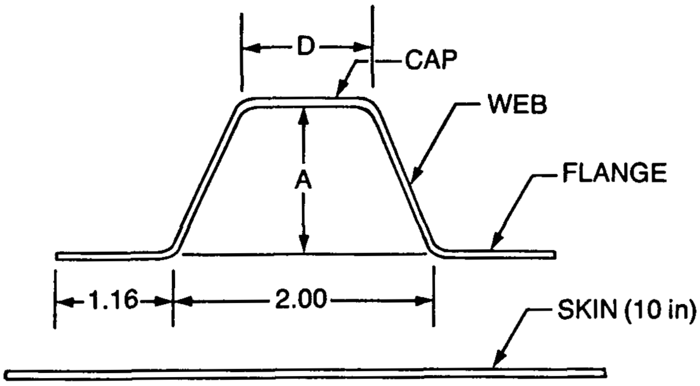
The existing requirements for flammability and fire protection of aircraft structure are designed to minimize the hazard to the occupants in the event that ignition of flammable fluids or vapors occurs. In addition, structural components exposed to heat, flames, or sparks should withstand these effects. The Federal Aviation Administration Composite Guidelines (AC 20-107) states that the use of composite structure should not decrease this existing level of safety (ref. 6.1.1-1). The concern is how new emerging requirements and guidelines may be modified in the future and what influence this will have on the use of materials presently considered for composite fuselage structures.

Technology voids that need to be addressed are (1) characterization of candidate material flammability properties, (2) design of fire protection systems, and (3) fire protection verification. Flammability properties that need to be characterized include ignition temperature, self-extinguishing characteristics, flame spread, and smoke content. The use of flame retardants and other fire protection systems will need to be considered during the fuselage design process. To ensure passenger safety, methods for determining the adequacy of fire protection for both exterior and interior surfaces of the fuselage shell will have to be evaluated, updated, and then used to verify fire protection systems.

6.1.2 Design Strain Levels

A basic issue for composite materials is to what strain levels can the fuselage structure be designed. Ultimate design strains are influenced by damage tolerance criteria in both tension and compression designed structure. Tension designed structure is controlled primarily by large area damage. Compression designed structure is controlled by either large area damage or residual strength after impact. The main concern for impact damage is what residual strength can be achieved considering minimum detectable damage sizes. The most direct design solution to produce damage tolerant structure is to lower the design strains.

The influence of design criteria on weight reduction has been quantified by analyzing a hat section stiffened laminate skin design (Concept 4), and an unstiffened honeycomb skin design (Concept 1) (see sec. 2.6). The study was performed in the crown and keel regions of the study section. In the study with the hat section stiffened laminate skin design, the nominal design strains were compared at 0.005 to 0.006 in/in tension and 0.004 to 0.005 in/in compression by changing skin and stringer laminate configurations. The geometry of the stringer cross section and stringer spacing was not varied. The results of this analysis are summarized in Figures 6.1-1 and 6.1-2. The weight reduction difference for the high strain designs, compared to the low strain designs, in the skin and stringers is approximately 72 pounds, as shown in Figure 6.1-3. This results in a reduction of an additional 2.8% of the total weight of the study section, based on a preliminary study section weight of 2590 pounds (fig. 3.2-1).



STA	D	A
1200	.90	.95
1340	.90	.95
1520	.90	.95
1701	.90	.95

DIMENSIONS IN INCHES

STA	LOAD, kip/in		LOW STRAIN DESIGN (2)				
	TEN	COMP	FLG/WEB	CAP	SKIN	STRAIN, in/in	$\bar{t}$ (1)
1200	5.0	−1.8	$(45/0/-45/\overline{90})_S$	$(45/0_2/-45/\overline{90})_S$	$(45/90/-45/0_2/\dot{0})_S$	.0045	.1161
1340	3.67	−1.33	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45/0_2)_S$	.0038	.1075
1520	2.50	− .90	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45/\dot{0})_S$	.0036	.0927
1701	1.95	− .90	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45/\dot{0})_S$	.0034	.0874

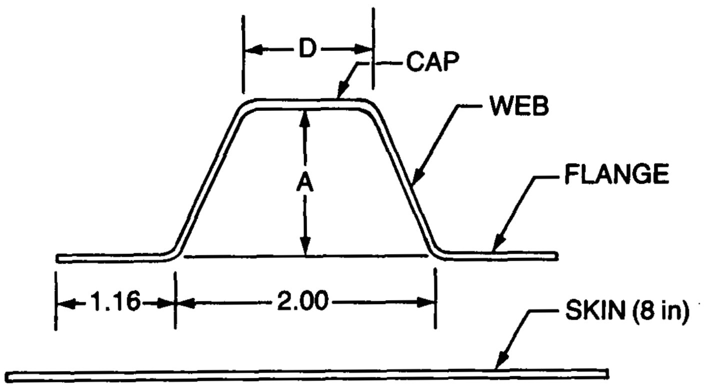
STA	LOAD, kip/in		HIGH STRAIN DESIGN (2)				
	TEN	COMP	FLG/WEB	CAP	SKIN	STRAIN, in/in	$\bar{t}$ (1)
1200	5.0	−1.8	$(45/0/-45/\overline{90})_S$	$(45/0_2/-45/\overline{90})_S$	$(45/90/-45/0/\dot{0})_S$	.0059	.1013
1340	3.67	−1.33	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45/0)_S$	.0053	.0927
1520	2.50	− .90	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45)_S$	.0059	.0779
1701	1.95	− .90	$(45/-45/90)_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45)_S$	.0060	.0766

(1) BASED ON 10-INCH STIFFENER SPACING

(2) THE CRITICAL DESIGN VALUES—BOTH STRAIN AND t—RESULT FROM TENSION LOADING AND ARE BASED ON ALLOWABLE TENSILE STRAINS OF .006 AND .005 FOR THE HIGH AND LOW STRAIN DESIGNS, RESPECTIVELY

ALL DIMENSIONS ARE IN INCHES. THE 0 deg FIBER DIRECTION IS NORMAL TO THE SKIN-STRINGER CROSS SECTION

Figure 6.1-1. Sensitivity of Crown Hat Laminate Configurations to Design Strain



STA	D	A
1200	.90	1.16
1340	.97	1.09
1520	1.61	.99
1701	1.15	.90

DIMENSIONS IN INCHES

STA	LOAD, kip/in	LOW STRAIN DESIGN (2)				
		FLG/WEB	CAP	SKIN	STRAIN, in/in	$\bar{t}$ (1)
1200	−5.50	$(45/0_4/-45/\overline{90})_S$	$(45/0_5/-45/\overline{90})_S$	$(45/90/-45/0_2)_S$	−.0037	.1522
1340	−3.56	$(45/0_2/-45/\overline{90})_S$	$(45/0_4/-45/\overline{90})_S$	$(45/90/-45/0/\overline{0})_S$	−.0035	.1257
1520	−2.00	$(45/0/-45/\overline{90})_S$	$(45/0_2/-45/\overline{90})_S$	$(45/90/-45/\overline{0})_S$	−.0033	.1009
1701	−1.50	$(45/0/-45/\overline{90})_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45)_S$	−.0029	.0953

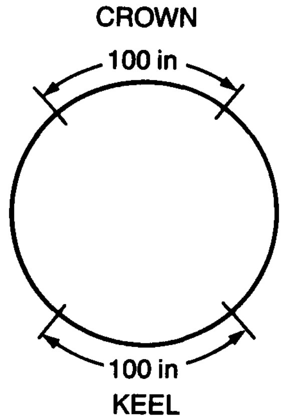
STA	LOAD, kip/in	HIGH STRAIN DESIGN (2)				
		FLG/WEB	CAP	SKIN	STRAIN, in/in	$\bar{t}$ (1)
1200	−5.50	$(45/0_3/-45/\overline{90})_S$	$(45/0_3/-45/\overline{90})_S$	$(45/90/-45/0_2)_S$	−.0044	.1400
1340	−3.56	$(45/0/-45/\overline{90})_S$	$(45/0_3/-45/\overline{90})_S$	$(45/90/-45/0/\overline{0})_S$	−.0043	.1150
1520	−2.00	$(45/-45/\overline{90})_S$	$(45/0_2/-45/\overline{90})_S$	$(45/90/-45/\overline{0})_S$	−.0043	.0923
1701	−1.50	$(45/-45/90)_S$	$(45/0/-45/\overline{90})_S$	$(45/90/-45)_S$	−.0042	.0872

(1) BASED ON 8-INCH STIFFENER SPACING

(2) BASED ON ALLOWABLE COMPRESSION STRAINS OF −.005 AND −.004 FOR THE HIGH AND LOW STRAIN DESIGNS, RESPECTIVELY

ALL DIMENSIONS ARE IN INCHES. THE 0 deg FIBER DIRECTION IS NORMAL TO THE SKIN-STRINGER CROSS SECTION

Figure 6.1-2. Sensitivity of Keel Hat Laminate Design Configurations to Design Strain



STUDY SECTION LENGTH
L = 540 in

HAT STIFFENED DESIGN WEIGHT STUDY [1] / LOCATION	LOW STRAIN DESIGN		HIGH STRAIN DESIGN		REDUCTION IN SMEARED THICKNESS $\Delta\bar{t}$, in	WEIGHT REDUCTION, lb [3]
	AVERAGE DESIGN STRAIN, in/in [2]	AVERAGE SMEARED THICKNESS $\bar{t}$, in	AVERAGE DESIGN STRAIN [2]	AVERAGE SMEARED THICKNESS $\bar{t}$, in		
CROWN	0.0038	0.1009	0.0058	0.0871	0.0138	42
KEEL	−0.0034	0.1185	−0.0043	0.1086	0.0099	30
					TOTAL WEIGHT REDUCTION	72

PRELIMINARY ESTIMATE OF STUDY SECTION WEIGHT (FROM FIGURE 3.2-1): 2590 lb
% WEIGHT REDUCTION OF HIGH STRAIN DESIGN: 72/2590 = 2.8%

[1] STRAIN AND SMEARED THICKNESSES FROM FIGURES 6.1-2 AND 6.1-3

[2] CRITICAL STRAIN IN CROWN IS IN TENSION
CRITICAL STRAIN IN KEEL IS IN COMPRESSION

[3] WEIGHT REDUCTIONS: (ϱ) (L) (100 in) $\Delta\bar{t}$

GR-EP DENSITY ϱ = 0.056 lb/in^3
STUDY SECTION LENGTH L = 540 INCHES
PANEL WIDTH = 100 INCHES

Figure 6.1-3. Sensitivity of Hat Stiffened Laminate Panel Weight to Design Strain

In the study with the honeycomb skin design, an assessment has been made of the possible weight changes resulting from varying minimum face sheet thickness requirements from five plies to four plies. The study was performed in the crown and keel regions. Descriptions of the initial and revised design configurations are shown in Figures 6.1-4 and 6.1-5. The main difference between the designs is that the revised design has, on the average, one less ply per face sheet. When extrapolated over 100 inches of the crown and keel and over 540 inches of the study section length, this reflects a weight reduction of approximately 75 pounds (see fig. 6.1-6). This is an additional 2.9% weight reduction to the total weight of the honeycomb design study section, based on the initial design (Concept 1).

This analysis demonstrates that weight benefits can be obtained by selecting materials that operate at increased strain levels. These benefits will need to be evaluated and traded against considerations such as (1) material toughness characteristics, and (2) design configuration. Materials that can operate at higher strain levels are generally less tough, and more prone to damage (see sec. 6.1.3). Structural details such as splices and cutout reinforcements can be designed to operate in a high strain field, but may require increased amounts of local reinforcement or load redistribution, which can reduce the weight benefits and increase fabrication cost.

6.1.3 Impact Damage

Impact may cause damage that varies from small internal delaminations to visually detectable skin punctures. The size of internal delaminations and the associated residual compression strength depends on impact energy, and structural response (ref. 6.1.3-1).

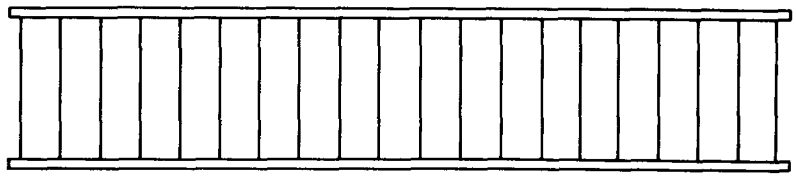
The significance of impact damage is directly proportional to the design strain. The higher the design strain, the greater the influence that impact damage has on the structure. The strength of postbuckled compression panels, as pointed out in Reference 6.1.3-2, will be influenced by impact damage due to the increase in surface strains caused by the buckle deformations.

The influence of impact damage can be reduced by several methods. The most direct way is to reduce the design strain. However, this leads directly to a heavier design. Another approach is to use a tougher material system that reduces the delamination area. However, tougher material systems may exhibit lower strengths in a hot wet environment. Increasing the resin content of the laminate has shown to produce an increase in load carrying capacity after impact. This approach also results in a heavier design.

Another method to minimize the effect of impact damage is to stitch through the thickness of the laminate. Stitching of the laminate with Kevlar thread provides transverse fibers that act to hold the laminate together and reduce the effect of the delaminations. These benefits were recently demonstrated in tests performed under NASA contract NAS1-16863 (ref. 6.1.3-3). Figure 6.1-7 summarizes the test results. This figure shows a reduction in delamination area and an increase in strain capacity for the stitched panels compared to the unstitched panels. In order for stitching to be viable, low cost methods need to be established for fabrication.

Based on this discussion, the following type of questions will need to be addressed in a composite fuselage technology development program:

- What is the level of impact damage that the panel must be tolerant to at design limit and ultimate loads?
- What are the geometric variables that improve impact resistance?
- What material and structural enhancements such as increasing resin content and stitching will provide a more weight efficient and cost effective structure?



- 4-lb/ft^3 FIBERGLASS HONEYCOMB CORE
- 20-in FRAME SPACING

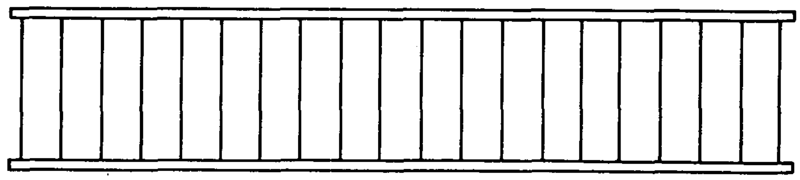
STA	LOAD, kip/in		INITIAL DESIGN			
	TEN	COMP	FACE SHEET LAYUP [1]	CORE HEIGHT C, in	STRAIN, in/in	$\bar{t}$ [2]
1200	5.0	-1.8	90/0/45/0/-45/0/90	.20	.0050	.1123
1340	3.67	-1.33	0/-45/90/45/0	.20	.0053	.0826
1520	2.50	-.90	0/-45/90/45/0	.15	.0036	.0802
1701	1.95	-.90	0/-45/90/45/0	.15	.0028	.0802

STA	LOAD, kip/in		REVISED DESIGN			
	TEN	COMP	FACE SHEET LAYUP [1]	CORE HEIGHT C, in	STRAIN, in/in	$\bar{t}$ [2]
1200	5.0	-1.8	0/45/0/90/-45/0	.20	.0052	.0971
1340	3.67	-1.33	0/-45/90/45/0	.18	.0053	.0814
1520	2.50	-.90	45/90/-45/0	.16	.0059	.0666
1701	1.95	-.90	45/90/-45/0	.16	.0047	.0666

[1] THE UPPER AND LOWER FACE SHEETS HAVE IDENTICAL LAYUPS

[2] SMEARED THICKNESS INCLUDES WEIGHT CONTRIBUTION FROM CORE.

Figure 6.1-4. Crown Honeycomb Design Configurations



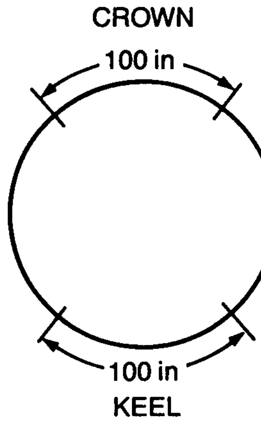
- 4-lb/ft^3 FIBERGLASS HONEYCOMB CORE
- 20-in FRAME SPACING

STA	LOAD, kip/in	INITIAL DESIGN			
		FACE SHEET LAYUP [1]	CORE HEIGHT C, in	STRAIN, in/in	$\bar{t}$ [2]
1200	-5.50	$(0/-45/90/45/\bar{0})_s$	.60	−.0048	.1588
1340	-3.51	90/0/45/0/-45/0/90	.50	−.0036	.1247
1520	-2.00	0/-45/90/45/0	.356	−.0029	.0887
1701	-1.50	0/-45/90/45/0	.296	−.0022	.0862

STA	LOAD, kip/in	REVISED DESIGN			
		FACE SHEET LAYUP [1]	CORE HEIGHT C, in	STRAIN, in/in	$\bar{t}$ [2]
1200	-5.50	(0/90/0/+45/-45/ 0/90/0)	.53	−.0044	.1403
1340	-3.51	90/0/45/0/-45/0/90	.37	−.0036	.1189
1520	-2.00	45/0/-45/90	.34	−.0046	.0732
1701	-1.50	45/0/-45/90	.26	−.0035	.0700

[1] THE UPPER AND LOWER FACE SHEETS HAVE IDENTICAL LAYUPS.
[2] SMEARED THICKNESS INCLUDES WEIGHT CONTRIBUTION FROM CORE.

Figure 6.1-5. Keel Honeycomb Design Configurations



STUDY SECTION LENGTH
L = 540 in

WEIGHT STUDY [1] LOCATION	AVERAGED SMEARED THICKNESS [2]		REDUCTION IN SMEARED THICKNESS $\Delta\bar{t}$, in	WEIGHT REDUCTION, lb [3]
	INITIAL DESIGN	REVISED DESIGN		
CROWN KEEL	0.0888 0.1146	0.0779 0.1006	0.0109 0.0140	33 42
			TOTAL WEIGHT REDUCTION	75

ESTIMATE OF INITIAL STUDY SECTION WEIGHT (FROM FIGURE 3.2-1): 2640 lb

% WEIGHT REDUCTION OF REVISED DESIGN: 75/2640 = 2.8%

[1] STRAIN AND SMEARED THICKNESSES FROM FIGURES 6.1-4 AND 6.1-5

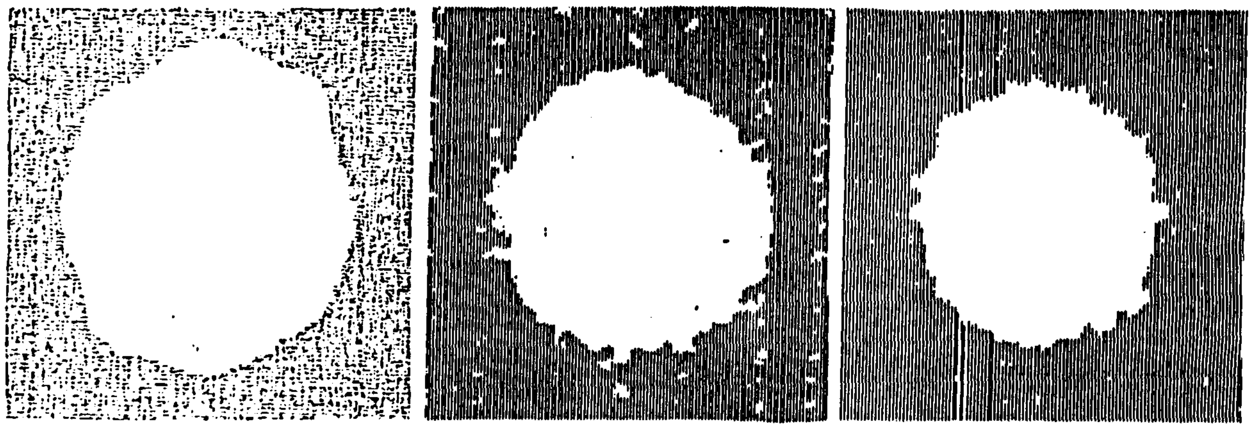
[2] SMEARED THICKNESSES INCLUDE WEIGHT CONTRIBUTION FROM CORE

[3] WEIGHT REDUCTION: $(\varrho)(L)(100)\,\Delta\bar{t}$

GR-EP DENSITY ϱ = 0.056 lb/in^2
STUDY SECTION LENGTH L = 540 INCHES
PANEL WIDTH 100 INCHES

Figure 6.1-6. Sensitivity of Honeycomb Skin Panel Weight to Design Criteria

UNSTITCHED	KEVLAR STITCHED—4 STITCHES/in ③	
	ROW SPACING = 0.5 in	ROW SPACING = .025 in



500 in-lb	500 in-lb	500 in-lb
25 ksi .00407 in/in	28.8 ksi .00428 in/in +15.2%	35.4 ksi .00544 in/in +41.6%

MATERIAL: HERCULES AS6/2220-3

① $(45/0/-45/90)_{5S}$ LAYUP; 5-in x 10-in x 40 PLIES; 0.5-in DIAMETER IMPACTOR
② GRADE 190 TAPE, 35% RESIN CONTENT
③ KEVLAR STITCHING IN AXIAL DIRECTION
FROM NASA CONTRACT NAS1-16863

Figure 6.1-7. Compression Strength After Impact of Stitched Panels—Coupon Evaluation

6.2 STRUCTURES

6.2.1 Pressure Damage Containment

The technical issue of pressure damage containment is a primary concern for the development of composite fuselage structure. The basis for this concern is due to (1) lack of analyses that model the structural behavior, (2) the lack of verification tests, and (3) the potential weight impact of having to add material to provide adequate damage tolerance.

The basic design criteria for pressure damage containment is that the pressure shell of the aircraft shall survive a 12-inch cut in any direction that may occur during a normal cruise flight condition. The energy of the damaging object shall be sufficient to completely sever a frame and/or stringer. The loading condition at the time of the incident is defined as a 1.0g flight load combined with a fail-safe pressure of 9.6 psi.

The damage tolerance capability of a plain sheet of graphite flat laminate has been established from center notch tests of coupons and panels. A review of industry data shows the results in Figure 6.2-1 (ref. 6.2.1-1). This data is based on T300 and AS-4 fibers, which are nominally 0.01 in/in strain to failure fibers. As shown in Figure 6.2-1, the largest damage that has been tested is 3.5 inches. If the curve is extrapolated to 12-inch damage, the resulting critical strain would be approximately 0.001 in/in.

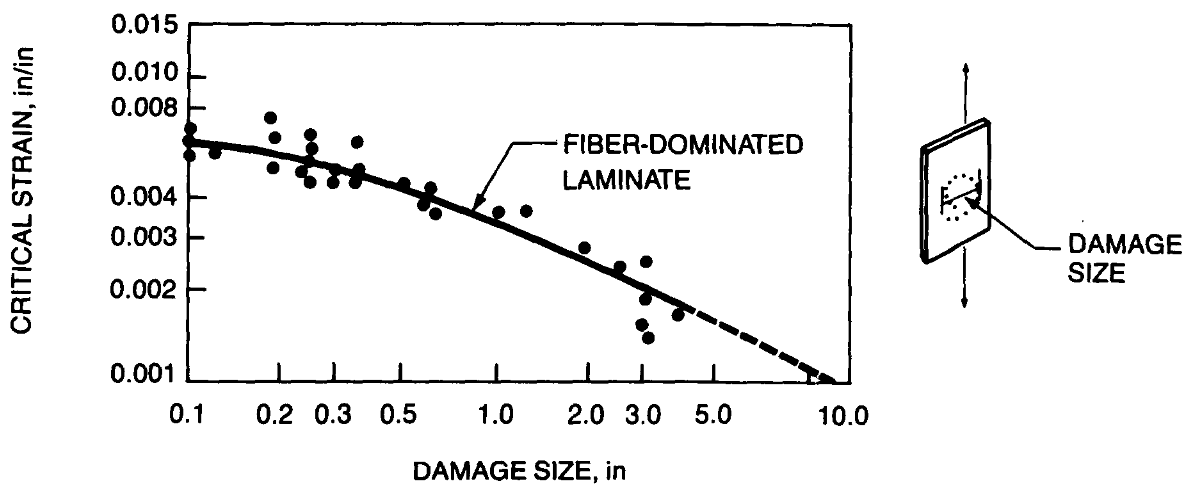
The fiber being considered for use in the fuselage development program has a nominal capability of 0.015 in/in strain to failure. Therefore, it is reasonable to assume that the curve in Figure 6.2-1 could be raised by a factor of 1.5. However, test results of open hole coupons with the higher strain to failure fibers have shown approximately a 1.4 factor improvement, which would result in a 0.0014 critical strain for 12-inch damage.

The fail-safe load condition of 1.0g is approximately 1/3 of the ultimate flight load condition. Therefore, a maximum allowable ultimate body bending tension design strain based on damage tolerance would be 0.0042 (3 x 0.0014) in/in disregarding the effects of temperature, moisture, and internal pressure. In a similar manner, the two-factor ultimate hoop pressure design strain would be 0.0026 in/in (0.0014 x 18.2/9.6).

Damage tolerance in fuselage structures can be achieved two ways. The first method is to size the basic skin to a strain level capable of withstanding the required damage size without tear straps. The second approach is to size the skin based on ultimate strength requirements, and then add tear straps as required to meet the damage containment requirement. To establish the weight difference between the two approaches, the fuselage skin with no tear straps was sized to contain a pressure load with a maximum design strain of 0.0014 in/in and a 50% correction factor for temperature, moisture, pressure, and curvature. This maximum strain value is based on a critical fiber strain of 0.015 in/in and a 12-inch damage size, as described in Section 2.2. The skin thicknesses that resulted from this study are compared in Figure 6.2-2 to the skin gages of the hat stiffened design with tear straps.

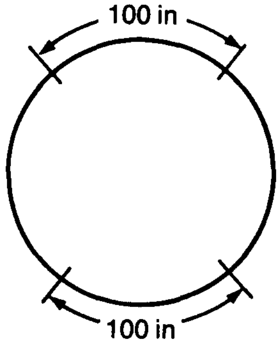
The weight of the skin designed to the low strain allowable is approximately 360 pounds heavier than the high strain skin and tear strap combination over the top and bottom 100 inches of the crown and keel, and over the 540 inches of the study section, as shown in Figure 6.2-2. The side regions are influenced by the window belt design considerations and are not included in this study.

These results indicate the importance of properly characterizing the pressure damage containment characteristics of composite fuselage structures and the usage of tear strap concepts. The flat panel fracture response data shown in Figure 6.2-1 needs to be expanded to include large discrete damages. The underlying assumptions used to develop the tear strap design curves presented in Section 2.3 need to be evaluated and verified by test. This should include the determination of characteristic dimension and critical strain data of applicable skin and tear strap laminate configurations. In addition, a correction factor (K) for temperature, moisture, pressure, and curvature needs to be determined. Strength reduction factors due to temperature and moisture need to be determined. Large damage fracture tests will need to be performed on curved panels subjected to pressure and then correlated with an analysis of the resultant out-of-plane peeling around the damage.



DATA BASED ON FLAT LAMINATE COUPONS WITH T300 AND AS-4 FIBERS
(CRITICAL FIBER STRAIN 0.010 in/in)

Figure 6.2-1. Typical Fracture Response of Flat Graphite Laminates



STUDY SECTION LENGTH: 540 in

STATION	CROWN		KEEL	
	SKIN GAGE WITHOUT TEAR STRAPS, in [1]	SKIN GAGE WITH TEAR STRAPS, in [2]	SKIN GAGE WITHOUT TEAR STRAPS, in [1]	SKIN GAGE WITH TEAR STRAPS, in [2]
1200	.140	.081	.131	.081
1340	.140	.074	.131	.074
1520	.140	.059	.124	.052
1701	.123	.059	.113	.044
AVERAGE SKIN GAGES	.136	.068	.125	.063

WT. OF SKIN GAGE WITHOUT TEAR STRAPS: (100) (540) (.136 + .125) (.056) ≅ 790 lb [3]
WT. OF SKIN GAGE WITH TEAR STRAPS: (100) (540) (.068 + .063) (.056) ≅ 400 lb [3]
WT OF TEAR STRAPS: 30 lb
WT DIFFERENCE BETWEEN TWO CONCEPTS: 790 − (400 + 30) = 360 lb

[1] SKIN GAGE BASED ON 0.0014 in/in TENSION ALLOWABLE SUBJECTED TO 9.6 psi, WITH A 0.50 CORRECTION FACTOR FOR TEMPERATURE, MOISTURE, PRESSURE, AND CURVATURE

[2] SKIN GAGE DEVELOPED FOR HAT STIFFENED LAMINATE CONCEPT 4

[3] GR-EP DENSITY = 0.056 lb/in^3

Figure 6.2-2. Skin Gage Requirements for Pressure Damage Tolerance

6.2.2 Postbuckled Structure

The issue of postbuckling strength is applicable only to laminate stiffened designs. Honeycomb structures are designed for buckling stability to 100% of design ultimate load (DUL). The issue of postbuckling strength for stiffened laminate panels includes the characterization of initial instability, out-of-plane skin deflections and associated skin-stringer disbonding.

A complete post-buckling panel analysis must include methods for predicting both initial instability and failure. The load level at which initial buckling of the skin is considered acceptable must be established based on a basic design criteria. This criteria will be influenced by factors such as aerodynamic smoothness requirements and a defined limitation to the number of times the structure will be allowed to buckle during one lifetime. There are several analysis programs available for predicting initial instability, such as PASCO (ref. 6.2.2-1, 6.2.2-2), STAGSC (ref. 6.2.2-3), and NASTRAN (ref. 6.2.2-4).

The strains in the skin and stringer elements in a postbuckled panel can be determined with finite element programs such as NASTRAN and STAGSC. However, modeling at the global level cannot be used to accurately predict panel failure since failures are typically controlled by the skin to stringer interface strength. Renieri and Garret (ref. 6.2.2-5) have developed some concepts for improving stringer-skin interface strength. These concepts, summarized in Figure 6.2-3, include three geometric tailoring concepts, a softening concept, and stitching. Renieri and Garret have demonstrated from finite element modeling of the interface that each of these concepts improves the static strength, as shown in Figure 6.2-4.

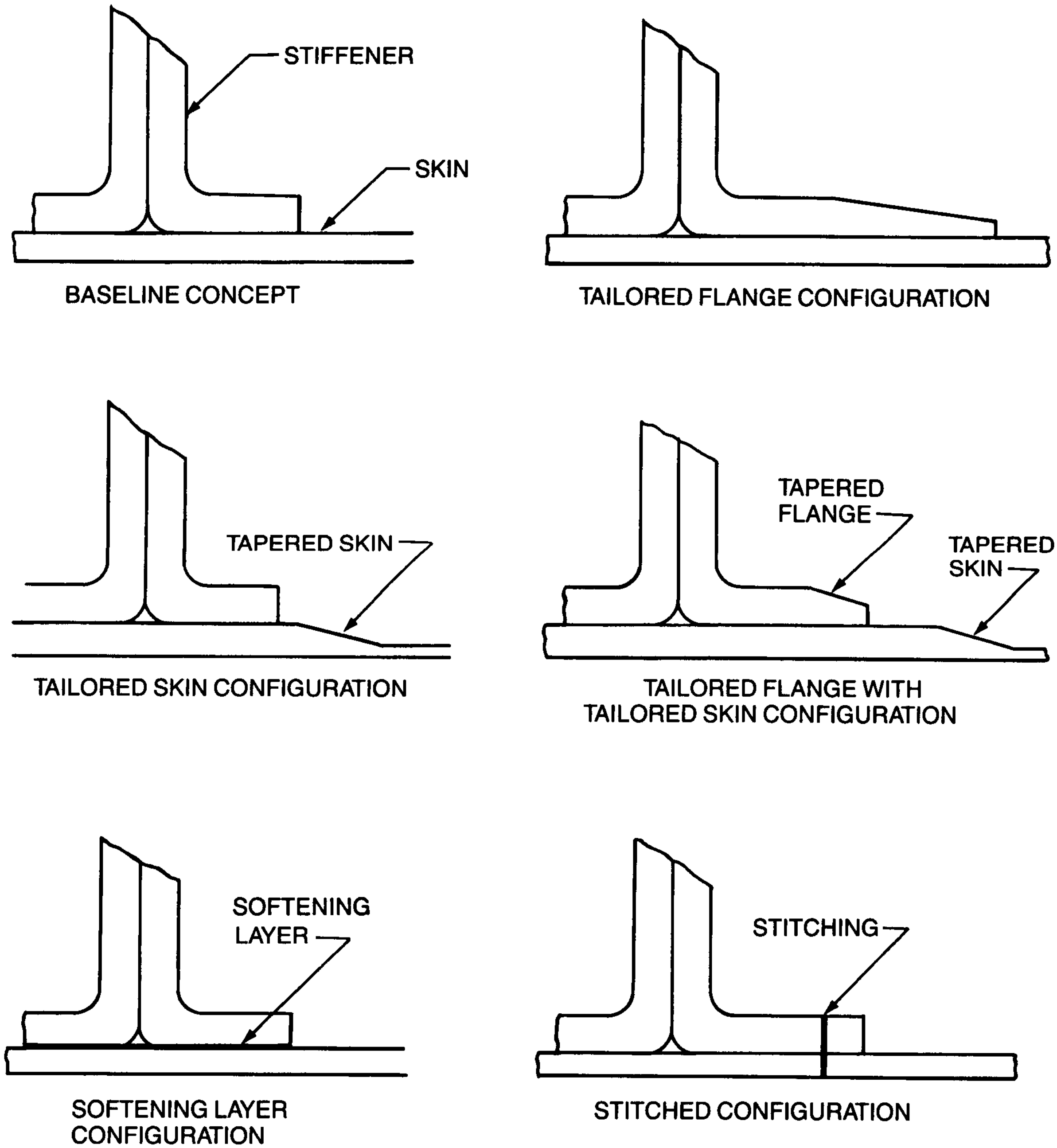
This discussion points out the need for an analytical procedure that will identify the loads at the skin-stringer interface in the postbuckled state. Testing needs to be performed to establish allowable design values for the interface strength. This data will be essential for determining methods for predicting ultimate strength of postbuckled skin-stringer panels.

6.2.3 Bolted Joints

The primary technology concern with bolted joints is how to use them effectively in fuselage splice design. Longitudinal and circumferential joints of a fuselage are predominantly biaxially loaded and may be subjected to high strain levels. Most existing bolted joint data, though, has been obtained from uniaxially loaded specimens.

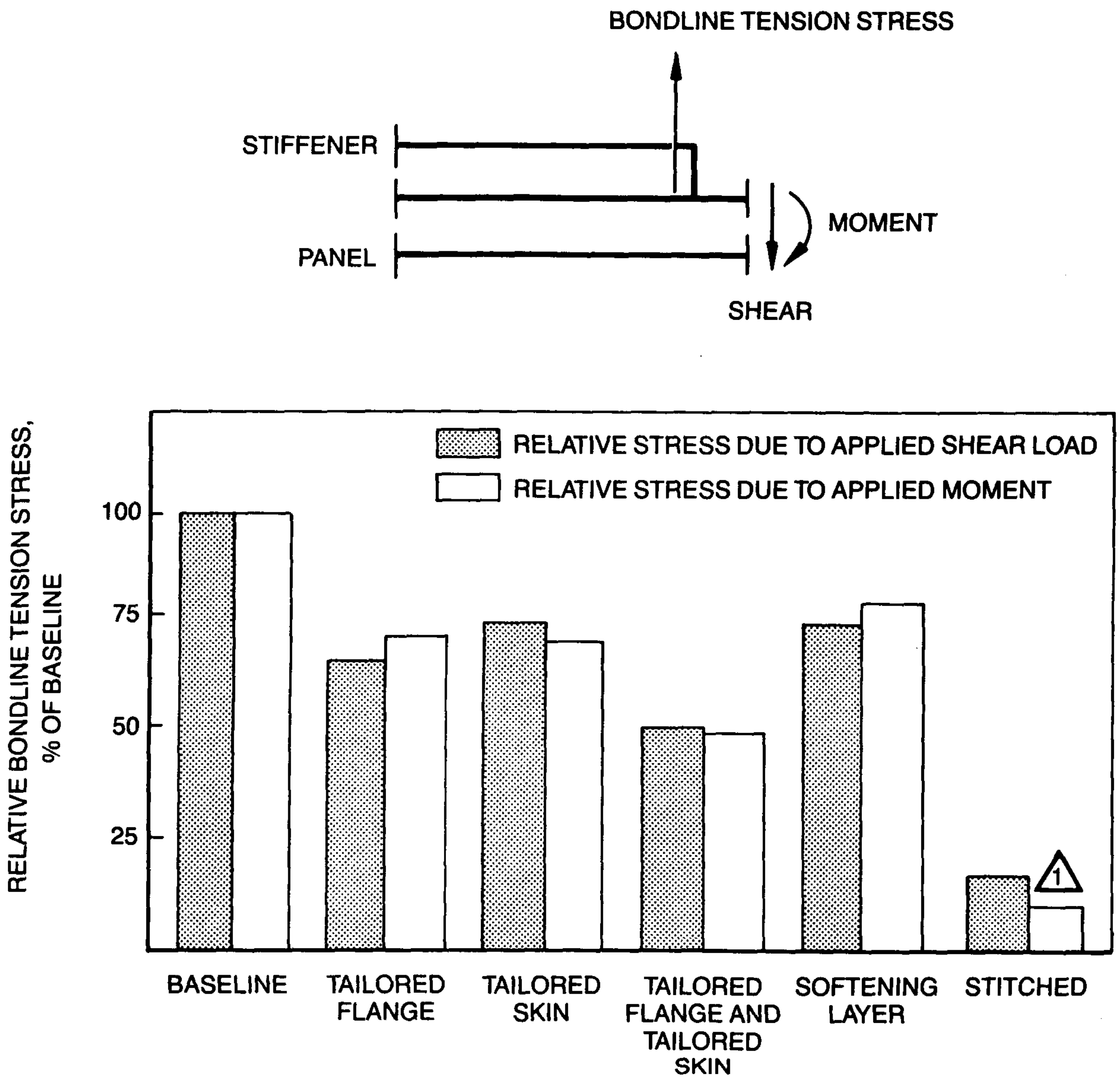
To assess the significance of bolted joint design, the longitudinal skin splices located at the crown and at the lower sides of the fuselage have been evaluated. The critical load conditions, summarized in Figure 6.2-5, include a maximum pressure condition and four flight maneuvers. The splice capability has been evaluated at Station 1200 using two skin laminates from hat section stiffened panel configurations designed to operate at tension strain allowables of 0.006 in/in and 0.004 in/in. These designs were previously defined in Section 6.1.1.

The Boeing version of the Air Force Bolted Joint Stress Field Model (BJSFM) (refs. 6.2.3-1 and 6.2.3-2) has been used to generate the bearing-bypass interaction curves shown in Figures 6.2-6 and 6.2-7. The shape of the curve is different for each laminate and angle between the bearing load and the far field bypass load. The bearing load (P_B) and load angle (α) result from the vector sum of the hoop and shear loads, as shown in Figure 6.2-5. The interaction curves are bounded by a 75 ksi bearing allowable. The longitudinal bypass stress is limited by strain allowables developed from uniaxial testing of laminate coupons.



FROM RENIERI AND GERRET, REFERENCE 6.2.2-5

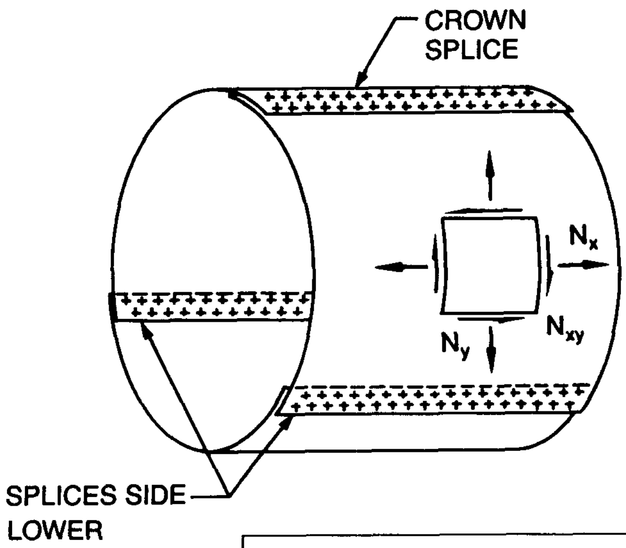
Figure 6.2-3. Skin-Stringer Interface Concepts



1 PEAK TENSILE STRESS ADJACENT TO STITCH

FROM RENIERI AND GERRET, REFERENCE 6.2.2-5

Figure 6.2-4. Finite Element Analysis of Skin-Stringer Interface Concepts

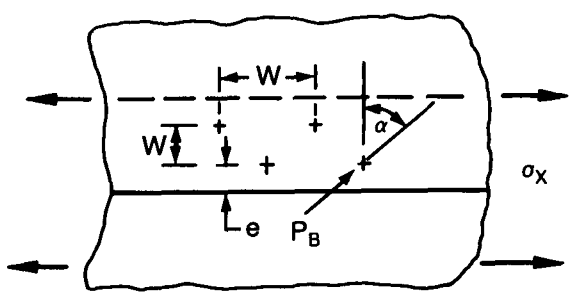


N_x: AXIAL LOAD

N_y: HOOP LOAD

N_{xy}: SHEAR LOAD

CRITICAL LOADINGS				
SPLICE LOCATION	LOAD CONDITION	LOAD (lb/in)		
		N_x	N_y	N_{xy}
CROWN	18.2 psi PRESSURE	650	1078	4
	BALANCED MANEUVER	3333	991	36
	LATERAL GUST	1749	881	98
	RUDDER MANEUVER	1239	891	247
LOWER SIDE	18.2 psi PRESSURE	462	1262	127
	ELEVATOR CHECK	604	1154	707

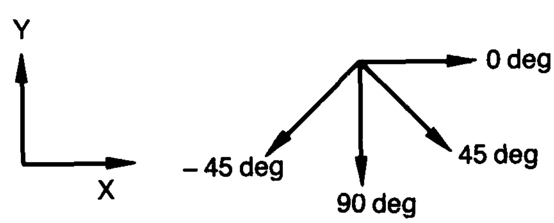


BOLT DIAMETER D = 3/16 in
W/D = 5
e/D = 2.5

LONGITUDINAL BYPASS STRESS
$\sigma_X = N_x/t_{LAM}$

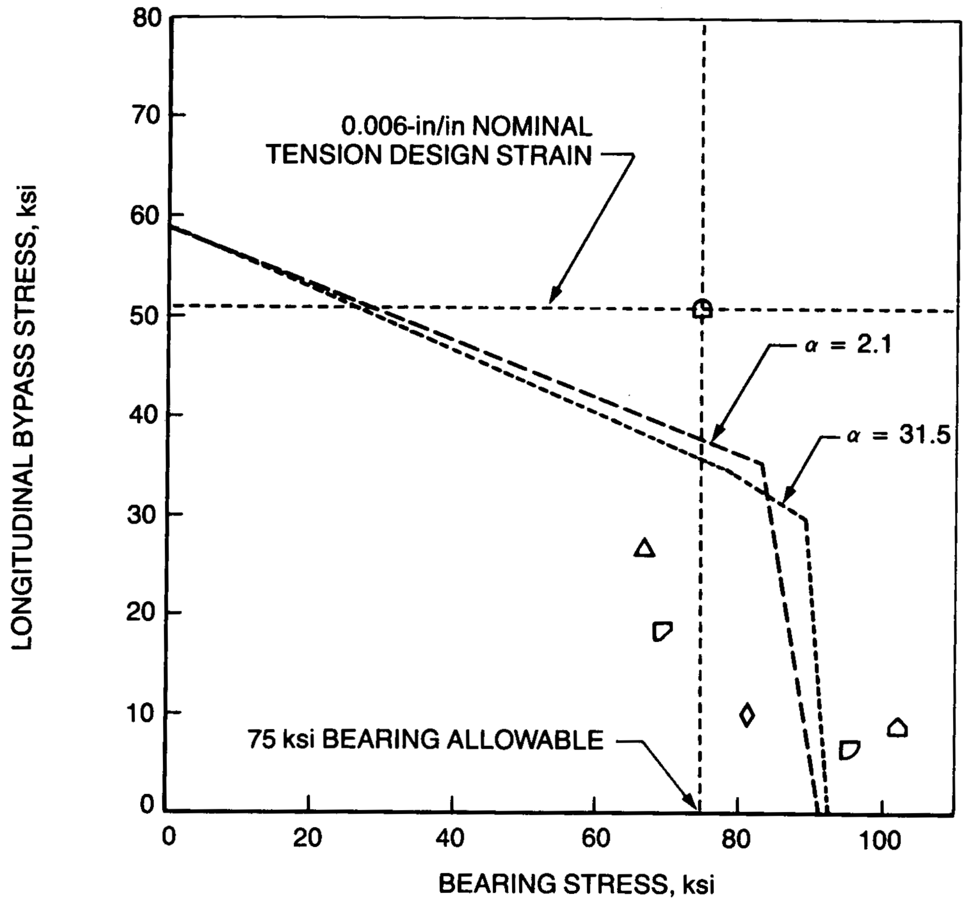
BEARING STRESS

$$\sigma_B = \frac{W}{D}\sqrt{N_y^2 + N_{XY}^2}\,/\,t_{LAM}$$



BEARING LOAD ANGLE
$\alpha = \tan^{-1}(N_{xy}/N_y)$

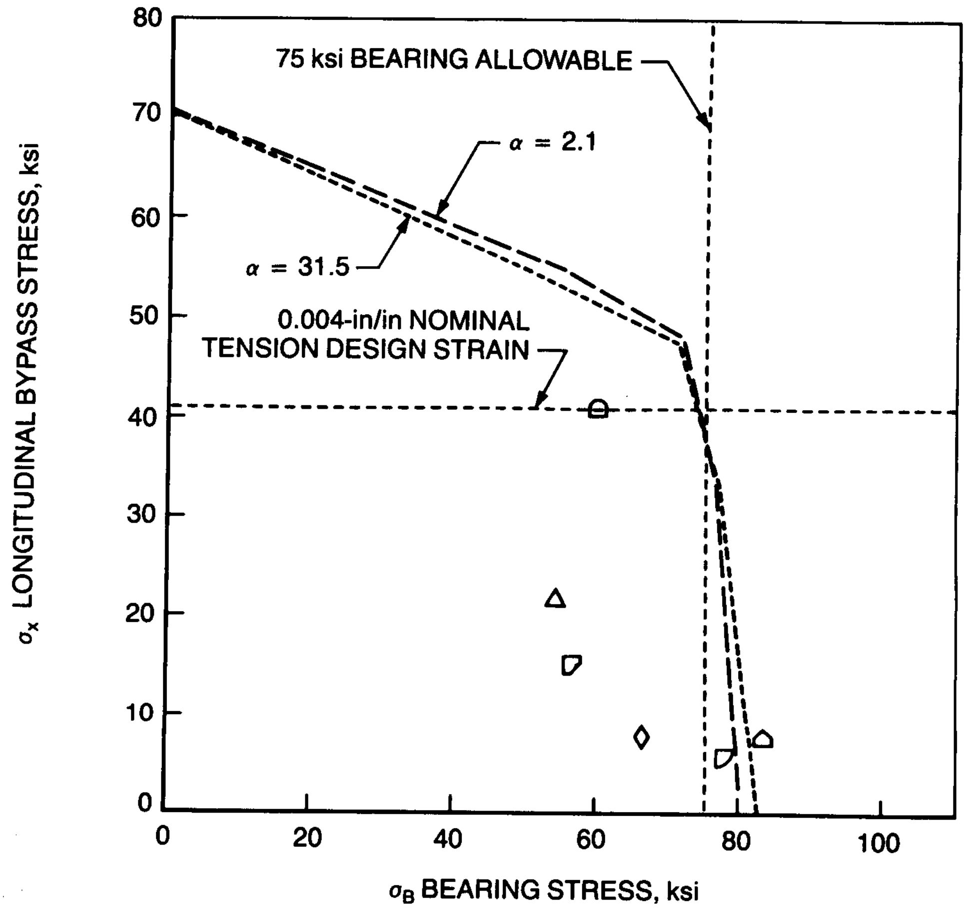
Figure 6.2-5. Longitudinal Splice Bolted Joint Parameters



SYMBOL	SPLICE LOCATION	LOAD CONDITION	BEARING LOAD ANGLE
◊	CROWN	18.2 psi PRESSURE	0.2
⌂		BALANCED MANEUVER	2.1
△		LATERAL GUST	6.3
▽		RUDDER MANEUVER	15.5
▽	LOWER SIDE	18.2 psi PRESSURE	0.2
⌂		ELEVATOR CHECK	31.5

SKIN LAMINATE $(45/90/-45/0/\bar{0})_S$ DEVELOPED FOR HAT STIFFENED LAMINATE CONFIGURATION WITH NOMINAL 0.006-in/in DESIGN STRAIN

Figure 6.2-6. Bearing-Bypass Interaction in Longitudinal Splice Design Based on 0.006-in/in Strain Allowable



SYMBOL	SPLICE LOCATION	LOAD CONDITION	BEARING LOAD ANGLE
◊	CROWN	18.2 psi PRESSURE	0.2
⌂		BALANCED MANEUVER	2.1
△		LATERAL GUST	6.3
▽		RUDDER MANEUVER	15.5
▽	LOWER SIDE	18.2 psi PRESSURE	0.2
⌂		ELEVATOR CHECK	31.5

SKIN LAMINATE $(45/90/-45/0/0/\bar{0})_S$ DEVELOPED FOR HAT STIFFENED LAMINATE CONFIGURATION WITH NOMINAL 0.004-in/in DESIGN STRAIN

Figure 6.2-7. Bearing-Bypass Interaction in Longitudinal Splice Design Based on 0.004-in/in Strain Allowable

The stress resultants for each of the load conditions and skin laminates are plotted with the allowable interaction curves in Figures 6.2-6 and 6.2-7. Without extra padding in the splice regions, the stress resultants for the skin laminate developed for the 0.006 in/in strain design exceed the allowable interaction curve in both crown and lower side regions (fig. 6.2-6).

In the crown, the splice design requires a pad-up of 14 plies in addition to the basic skin laminate, whereas the lower side splice requires four plies of pad-up in addition to the basic skin. A pad-up of 14 plies extrapolated over the full length of the study section results in a weight penalty of approximately 12 pounds, or roughly 0.5% of the total section weight. The pad-up material can effectively carry part of the bypass loading, thus allowing the material in the adjoining skin and stiffeners to be reduced. Concerns are that the pad-up in the crown splice may create unacceptable eccentricities, and that it would increase fabrication costs.

An alternative to padding-up the crown splice is to reduce the allowable bypass design strain. By reducing the maximum tension strain in the crown region to 0.004 in/in, the resulting crown skin splice is within allowable limits without further pad-up, as shown in Figure 6.2-7. The corresponding stress resultants in the lower skin are outside the allowable limits. The lower side splice for this design requires only two plies in addition to the basic skin. The weight penalty associated with the configuration designed to 0.004 in/in is approximately 72 pounds, as described in Section 6.1.1.

Another design solution would be to move the splice off of the top to a lower position on the crown. By doing this, the extensional bypass strains are reduced, thus allowing higher bearing loads. The fuselage would then be fabricated using four major panel segments instead of three, which would result in additional assembly costs.

Technology voids that need to be addressed are to (1) obtain biaxial bolted joint strength allowable data, and (2) perform cost-weight trade studies on splice design. Tests should be performed to determine the strength capability of biaxially loaded joints. This data should be used to verify analytical bearing-bypass stress interaction plots similar to those shown in Figures 6.2-6 and 6.2-7. Detailed cost-weight tradeoff studies need to be performed in order to establish optimum splice design configurations. These studies should include consideration of design strain level, splice location, splice geometry, and load path redistribution.

6.2.4 Cutouts

The primary concern for large cutouts is how to build up the reinforcement material around the cutouts without creating severe interlaminar stresses.

The material around cutouts needs to be designed in a manner that leads to an effective load transfer around the cutout. The effectiveness of the design depends on the ability of the material to transfer load, through shear, around the cutout. If the transition is abrupt, the interlaminar stresses in this area will be high and possibly result in delamination. In order to reduce the potential for delaminations, designs need to be developed that minimize interlaminar stresses in the transition area around the cutout.

A pad-up concept for reinforcing cutouts is shown in Figure 6.2-8. The pad-up is made by progressively adding plies to the skin. Cutout designs need to be analyzed to determine if the stresses in the pad-up region are less than the allowables. Interlaminar strength allowables do not currently exist and need to be established.

An additional concern in large cutout regions is that most laminate design values have been obtained from uniaxial coupon testing and very minimal design values are available for laminates in a combined stress field such as around cutouts.

6.2.5 Impact Dynamics

The main concern for impact dynamics is whether or not existing FAA recognized design load factors used in structural analysis for emergency landing load conditions will be suitable for composite fuselage design. Analytical models that contain load response of composite elements will have to be developed, analyzed, and evaluated. These composite analyses must be compared to similar analyses performed for aluminum structure. Based on these analyses, the suitability of using the existing emergency landing condition load factors and design and analysis methods to design composite components will need to be determined.

The FAA requirements for emergency landing conditions summarize the structural requirements necessary for passenger safety. The general requirements from Section 25.561 of FAR 25 (Ref. 6.2.5-1) are quoted below:

"25.561 General

(a) The airplane, although it may be damaged in emergency landing conditions on land or water, must be designed as prescribed in this section to protect each occupant under these conditions.

(b) The structure must be designed to give each occupant every reasonable chance of escaping serious injury in a minor crash landing when -

(1) Proper use is made of seats, belts, and all other safety design provisions;
(2) The wheels are retracted (where applicable); and
(3) The occupant experiences the following ultimate inertia forces acting separately relative to the surrounding structure:

(i) Upward - 2.0g
(ii) Forward - 9.0g
(iii) Sideward - 1.5g
(iv) Downward - 4.5g, or any lesser force that will not be exceeded when the airplane absorbs the landing loads resulting from impact with an ultimate descent velocity of five f.p.s. at design landing weight.

(c) The supporting structure must be designed to restrain, under all loads up to those specified in paragraph (b)(3) of this section, each item of mass that could injure an occupant if it came loose in a minor crash landing."

All passenger payload support structure must be designed to withstand the inertia loads described above. These design criteria should be considered for any materials used in construction of the fuselage and passenger support structure.

The response of a fuselage structure to a dynamic impact depends on the energy absorption characteristics of the material and the response of the design configuration. Due to plasticity, aluminum materials at the test coupon level absorb more energy than graphite-epoxy materials that are elastic to failure. However, the fuselage design configuration has a significant effect on energy absorption. This has been demonstrated through impact drop tests of a forward and aft section of an aluminum 707 body section, performed by NASA (ref. 6.2.5-2). Accelerometers were used to monitor inertia forces transferred to structural floor details and "dummy" passengers in seating areas. The failure modes of the fore and aft sections are shown in Figure 6.2-9.

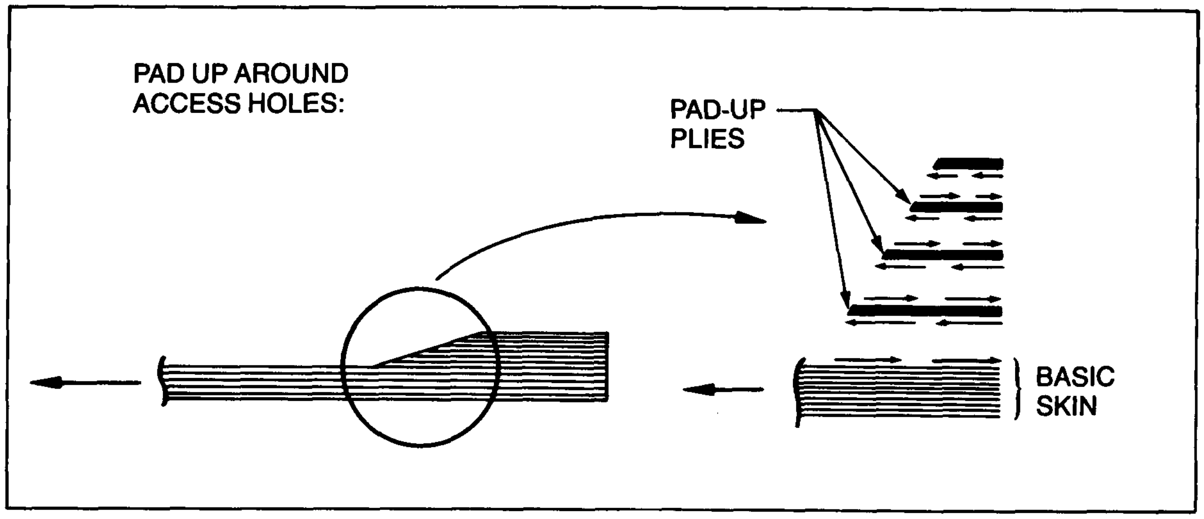


Figure 6.2-8. Shear Transfer in Pad-Up Region Around Cutouts

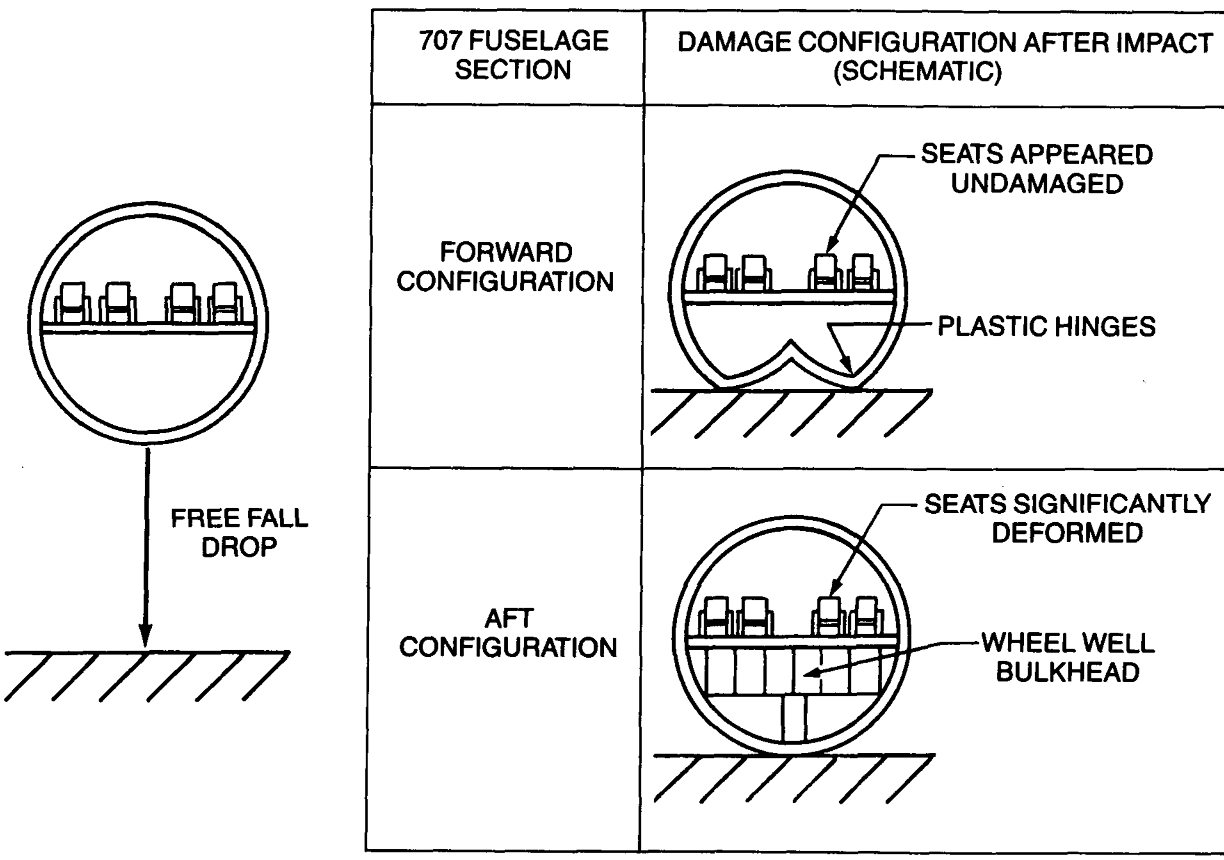


Figure 6.2-9. Influence of Configuration in Impact Dynamics Tests of 707 Fuselage

The seat structures in the rear section contained definite visual deformations, whereas the seats in the forward section appeared undamaged. The structural configuration below the floor in the aft fuselage is much more rigid than in the forward section. The energy imparted to the keel beam in this section is transferred up through the rigid structure to the passenger area, thus creating high inertia forces. The configuration of the forward section developed plastic hinges on the lower sides during impact. As a result, considerable energy absorption occurred due to structural distortions.

In order for a new fuselage configuration to be viable, the structure must exhibit equivalent passenger protection during a similar impact scenario as currently certified fuselage structures. Dynamic analyses are being developed that characterize the response of metal fuselage structures. This work is being funded by NASA contract NAS1-16076 and utilizes the NASA sponsored analysis program, DYCAST, developed by Grumman Aerospace. The analysis program DYCAST has been shown to provide reasonable correlation with the dynamic response exhibited in metal fuselage drop tests. In order for the program DYCAST to be applicable to composites, composite analysis elements need to be developed that account for their reaction to load and fracture response. The application of composite configuration constraints imposed by manufacturing, systems, and structures will influence fuselage response and need to be included in the analysis.

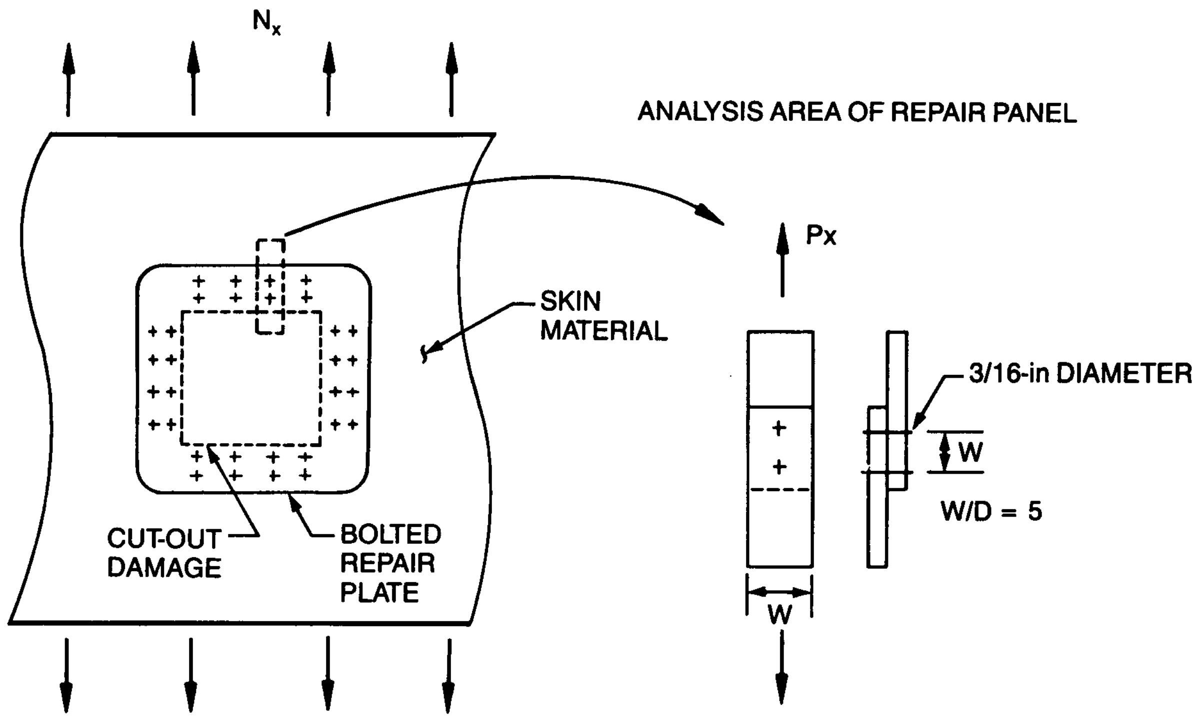
6.2.6 Repair

The main issues that relate to repair are associated with the trade-offs between structural repair capability and repaired part performance. In a typical composite repair procedure, the damaged area is removed and then replaced with a precured part. The repair part may be applied using mechanical fasteners, adhesives, or a combination of both. In order to simplify repair procedures and minimize airplane downtime, the most efficient method of attaching the repair part is to use mechanical fasteners. Methods that employ fastened repair plates require little specialized training for airline repair personnel. The alternative to fastened repair methods is to employ bonded repairs, which often require specialized training and facilities.

The feasibility of using the bolted repair approach has been evaluated by analyzing a repair located in a skin section designed to a tension strain of 0.006 in/in (shown in fig. 6.2-10). The uniaxial loading considered for this analysis is representative of the fuselage crown area during a balanced maneuver. For a double fastener joint, loading creates a 108 ksi bearing stress in the skin to repair joint, based on load transfer through the joint. This calculation does not include the effects of load redistribution around the cutout. In order to reduce the bearing stresses in the joint to an allowable 75 ksi bearing, the skin in the area of the repair requires four additional layers. The application of doublers around the damage cutout normally requires a bonding operation that is time consuming and costly. An alternative approach is to use a combination of bolted and bonded joining mechanisms for attaching the repair doublers. The methods for this type of repair have been developed and verified for the 737 stabilizer and are described in Reference 6.2.6-1.

In comparison, if repairs were made in a skin section designed to a tension strain of 0.004 in/in, the bearing stress would be reduced to 72 ksi (108 x 0.004/0.006), which is below the allowable of 75 ksi. This comparison points out that repairs could be made without the requirement of having to bond in additional plies; thus, repairs would be less costly in structures that are designed to the lower ultimate strains.

The principal repair subjects that need to be addressed in a composite fuselage technology development program are (1) to perform design-repair cost trade studies, and (2) to verify repair adequacy. Trade studies between repair cost and design strain level should be performed over all areas of the fuselage, with specific attention given to areas most frequently damaged. Consideration should be given to fuselage design concepts that will permit low cost repairs with a minimum of airplane downtime. Once established, repair procedures will need to be structurally evaluated. Tests should be performed on repaired shell structure components subjected to critical loads with adverse environmental conditions.



ASSUMPTIONS

(1) SKIN DESIGNED TO 0.006-in/in TENSION STRAIN LEVEL RESULTING FROM UNIAXIAL LOADING Nx
(2) LOAD REDISTRIBUTION AROUND HOLE NOT CONSIDERED
(3) REPAIR PANEL AND SKIN CONFIGURATION:

QUASI-ISOTROPIC LAMINATE: $(45/90/-45/0)_S$
$t = 0.0592$ in $E = 7.2$ MSI

(4) BEARING ALLOWABLE: $\sigma_{ALLOW} = 75$ ksi

ANALYSIS:

END LOAD Nx AT 0.006-in/in STRAIN: $Nx = \varepsilon tE = 2557.4$ lb/in
CONSIDER STRIP OF MATERIAL OF WIDTH $W = 5(D) = 0.9375$ in
LOAD IN STRIP $Px = N_x W = 2397.6$ lb
WITH LOAD P SHARED EVENLY BETWEEN TWO BOLTS
BEARING STRESS: $\sigma_B = Px/2tD = 108$ ksi*
*UNACCEPTABLE BEARING STRESS

MINIMUM GAGE TO REDUCE BEARING:

$t_{min} = Px/2D\,\sigma_{ALLOW} = 2397.6/(2)(0.1875)(75000) = 0.0852$ in (12 PLIES)

Figure 6.2-10. Bolted Repair Study

6.3 SYSTEMS

Design and implementation of systems within a graphite-epoxy composite fuselage airplane will require a significant change in design and analysis ground rules. The structure and substructure, being no longer either good electrical or thermal conductors, may not be employed as system elements (e.g., electrical ground return paths, heat sinks) nor do they provide the same protective and isolated environment against deleterious induced effects from atmospheric electrical hazards.

A list of technology issues pertaining to systems development is shown in Figure 6.3-1. Possible weight penalties, shown in Figure 6.3-1, attributable to each of the technology issues were estimated to total 1170 pounds including composite wing system technology developments. The proposed systems solutions are shown in Figure 6.3-2. If research efforts specifically directed towards the unique systems requirements in the fuselage were to be integrated during fuselage design development, then the weight penalty could be reduced to 550 pounds. Each of the systems technology issues is discussed below.

Thermal analyses will need to be developed for a composite fuselage in order to determine insulation requirements. It is anticipated that enough insulation can be supplied using current materials without any weight impact.

6.3.1 Fuselage Lightning Protection (Direct Effects)

The event of being hit by a lightning strike must be considered in fuselage design. A lightning strike can cause significant damage at the point of attachment and induces a current through the fuselage that can lead to sparking and heating of joints (fig. 6.3-3). Since the electrical conductivity of graphite-epoxy is significantly less than that of aluminum, the energy transferred by a lightning strike does not dissipate as easily as it would in an aluminum structure. The most direct effect of this is that the localized structure around the lightning attachment point will be subjected to a high impulse of energy that can cause severe heating and degradation of the structure.

In order to dissipate the strike energy, the conductivity of a graphite-epoxy fuselage shell must be increased. Methods for increasing the conductivity of a composite laminate include the application of conductive paints or primers, metal meshes and sheeting, and metal fibers woven through the laminate. In addition, metal coatings, such as nickel, can be electroplated to the graphite fibers before impregnation with the epoxy matrix.

The wire screen and foil concepts have been verified by Boeing in an Air Force contract, Reference 6.3.1-1, and the nickel coated fiber concept has been verified by ongoing Boeing development programs. In addition to being lighter, nickel coated fibers have better galvanic compatibility with graphite fiber than aluminum. It is anticipated that new materials will be developed for both composite wing and fuselage structures and these materials will provide more weight efficient methods of increasing the shell conductivity. Tests must be conducted for each candidate composite design in order to determine the extent of damage due to a lightning strike. Parameters that need to be addressed include laminate orientation and thickness, and the amount and type of paint or coating on the outside surface. The differences between skin stringer and honeycomb panels also need to be characterized.

The lightning protection system will provide some degree of protection to electrical/electronic systems from lightning-induced transients. The degree of such protection and the consequent reduced level of transient hardening required of systems components and wiring will have to be determined by analysis and verified by test.

SYSTEMS TECHNOLOGY ISSUE	TECHNOLOGY SOLUTIONS DEVELOPED AFTER STRUCTURAL DESIGN IS COMPLETE	WEIGHT PENALTY, lb [1]	TECHNOLOGY SOLUTIONS INTEGRATED DURING DESIGN PROCESS	WEIGHT PENALTY, lb [1]
1. FUSELAGE LIGHTNING PROTECTION	GR-EP FIBERS COATED WITH NICKEL IN SURFACE PLIES	300	POTENTIAL OF FUTURE MATERIAL DEVELOPMENTS	100
2. ELECTRICAL CIRCUIT RETURN	DEDICATED ELECTRICAL CIRCUIT RETURN FOR 90% OF WIRES	120	SAME	120
3. ELECTRICAL/ELECTRONIC BAYS SHIELDING	ENCLOSES BAYS WITH ELECTRO-MAGNETIC SHIELDING	300	ELECTROMAGNETIC HARDENING OF ELECTRICAL CIRCUITS. CONNECTORS AND CABLES SHIELDED	150
4. FLIGHT DECK EQUIPMENT PROTECTION	ALUMINUM FOIL OVER ENTIRE INNER COMPOSITE SURFACE	70	UTILIZE INDIRECT BENEFITS FROM OTHER SYSTEMS PROTECTON HARDWARE	60
5. SIGNAL WIRE AND POWER DISTRIBUTION PROTECTION	FIBER OPTICS REPLACING SIGNAL WIRE: OVERBRAID ON POWER WIRES	0	SAME	0
6. PERSONNEL PROTECTION	METAL FLOOR IN PASSENGER AND COCKPIT AREAS, TIED TO STRUCTURE ONCE PER FOOT	100	TOTAL ELECTRO-ISOLATION OF PASSENGER FLOOR AREAS FROM STRUCTURE	20
7. ACOUSTIC TRANSMISSION	INSULATION MASS	280	ADVANCED DESIGN OF COMPOSITE FUSELAGE COMBINING BOTH STRUCTURAL AND ACOUSTIC REQUIREMENTS	100
	TOTAL	1170	[2]	550

[1] WEIGHT PENALTY RELATIVE TO ALUMINUM FUSELAGE

[2] TECHNOLOGY ISSUES CONSIDERED SEPARATELY
WEIGHT PENALTY WILL BE REDUCED AFTER INTERACTIVE EFFECTS CONSIDERED

Figure 6.3-1. Weight Penalties Associated With Composite Fuselage System Technology Issues

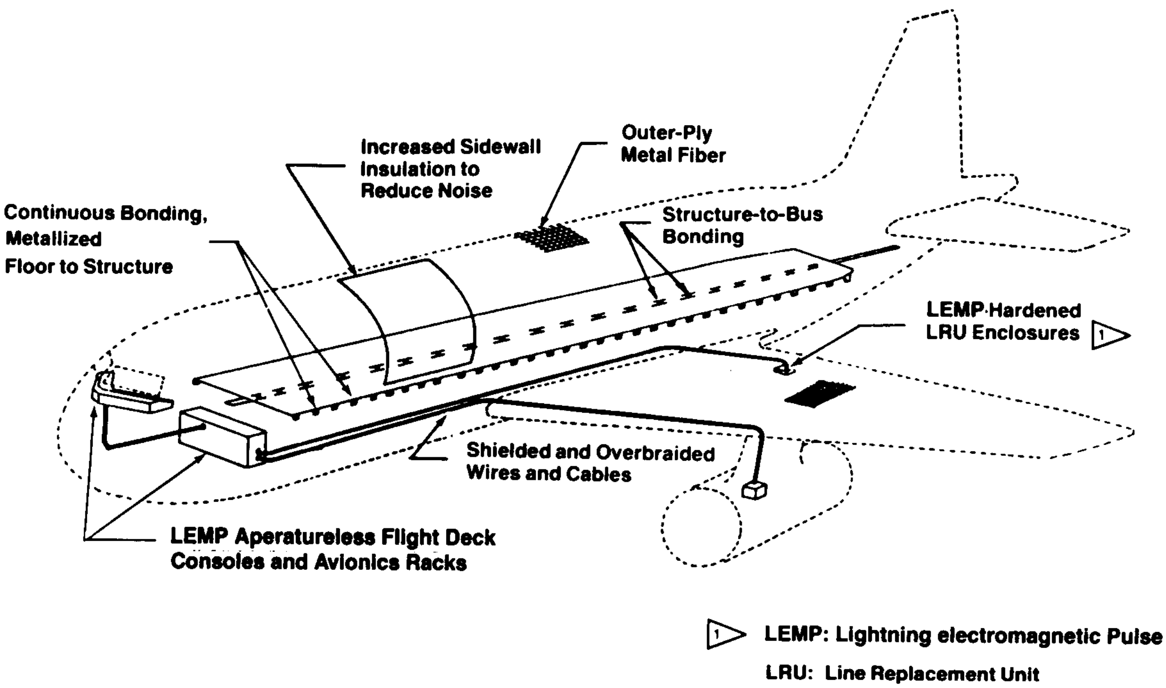


Figure 6.3-2. Solutions to Systems Technology Issues

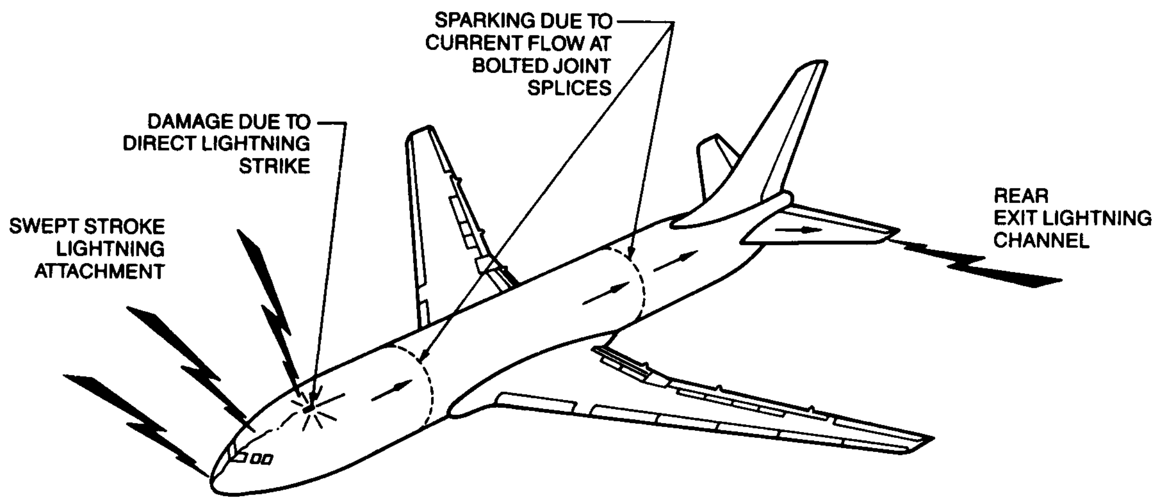


Figure 6.3-3. Direct Lightning Strike Effects

Typically, bolted structural joints adequately carry lightning currents without detrimental effects. The requirements for structural loads generally dictate numerous mechanical fasteners and thereby provide multiple paths for lightning current flow. It is possible, though, that sparking or heating may occur across a structural joint during a current flow (fig. 6.3-3). Candidate joint configurations must be evaluated through test at current levels characteristic during a lightning strike. If there is sparking and if heating is so severe that resin strength is reduced, the joints will have to be modified. The design solution may include such alternatives as bare metallic fasteners or conductive materials added to the joint.

6.3.2 Electrical Circuit Returns

Due to the high resistivity of graphite-epoxy, the composite fuselage structure cannot be employed as an electrical or fault circuit current return path. Provision of both of these functions for systems in a composite fuselage is possible by means of grounding buses or bars, metal conduits or raceways serving both shielding and grounding, or ground plane conductors embedded in the structure.

The electrical circuit returns could be assured by the addition of dedicated wires for each circuit. Using the Boeing 757 as a baseline for electrical wiring runs and weights and assuming that 75% of the circuits would require such dedicated returns, the weight penalty associated would be approximately 120 pounds.

6.3.3 Electrical/Electronic Equipment Bays Shielding

Current metal airplane electrical/electronic equipment bays are open racks containing line replaceable units (LRU) and enclosures providing adequate electro-magnetic induced (EMI) shielding between systems components and between systems and environment. The composite fuselage airplane will require some redesign of the equipment bays to account for the increased severity of the environment. One method for achieving a benign operating environment for the equipment is to enclose the racks with lightweight electromagnetic shielding.

With analysis and design attention to hardening the LRU cases, to shielding of systems interconnection cables and surge isolation of flight-critical systems functions, and to a cooling system for the LRUs that allows fewer apertures for EMI, it is estimated that the solution weight penalty could be reduced.

6.3.4 Flight Deck Equipment Protection

In order to maintain the integrity of avionic equipment in the flight deck during lightning strike, significant electromagnetic shielding is needed. In an airplane with an aluminum fuselage, flight deck shielding is supplied by the conductivity of the surrounding structure. In a composite airplane, proper shielding can be obtained by adding aluminum foil to the flight deck surrounding. For example, with aluminum foil adhesively bonded to the entire inner surface of a flight deck, a substantial degree of shielding would be achieved for electromagnetic frequencies below 10 MHz, where the drop-off commences in shielding effectiveness of graphite-epoxy composites. It is anticipated that trade-off combinations of LRU hardening and enhanced direct effects protection benefits against induced effects could reduce this weight penalty.

6.3.5 Signal Wires and Power Distribution

Special electromagnetic shielding is needed around all wires exterior to the electrical/electronic bays and flight deck. The incidence of an induced transient in the control wiring could be very serious, especially if flight surfaces are controlled by electrical systems. A potential protection method chosen for the wiring in an airplane similar in size to the Boeing 757 entails a metal overbraid on 85% of wires and cables exterior to the equipment bays and flight deck. However, further developments brought about by the need to minimize weight penalties for both composite wing and fuselage structures need to be explored. One solution would be the application of EMI-impervious fiber optics signal transmission lines between LRUs. At present, it is estimated that about 60% of signal wires could be replaced by the lighter fiberoptic buses, so that even with overbraid addition employed for remaining wire and cable run protection there would be no weight penalty.

6.3.6 Personnel Protection

Lightning currents through a graphite-epoxy composite structure can produce a large voltage difference along the structure between the entry and exit attachment points. Besides creating a potential voltage problem for equipment inside the vehicle, this poses a potential hazard for passengers and crew, as depicted in Figure 6.3-4.

If a metallic floor is used for flight deck and passenger cabin and is electrically bonded to the fuselage shell at only one point near the forward end of the airplane, the entire length of the metallic floor will stay at the same electrical potential as the front end of structure, where the two are bonded. The lightning currents down the fuselage can produce voltage differences of 10,000 volts or greater between the forward and aft end of the fuselage. If a person in contact with the metal floor were to touch the fuselage structure, the electrical circuit would be completed, causing a hazardous shock.

A relatively simple solution would be to add metal grounding strips to the floor in the passenger and flight deck areas. In addition, the metal strips would have to be attached to the shell structure regularly along the fuselage length. A more weight efficient solution is to provide total electro-isolation of passenger floor areas from the shell structure by means of high dielectric material.

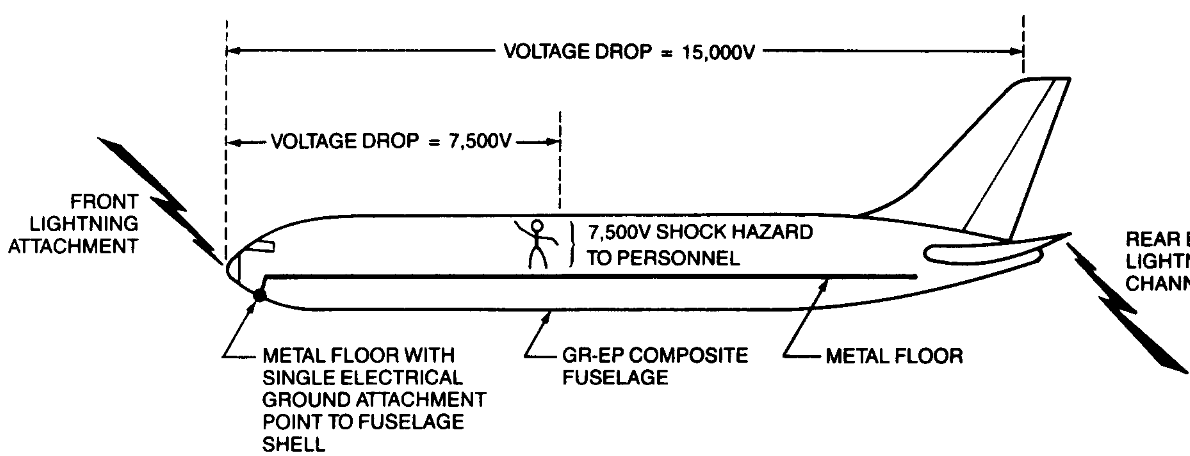


Figure 6.3-4. Potential Shock Hazard to Personnel Due to Lightning Strike

C - 2

6.3.7 Noise Attenuation

The noise in the interior cabin needs to be maintained at levels acceptable for passenger comfort. The ability of the fuselage to attenuate noise is influenced by the weight and structural configuration of the fuselage shell, passenger and cargo supporting structure, and interior panels.

The noise attenuation characteristics of a representative composite fuselage structure with laminate skin and I-stiffeners has been compared to that of a baseline aluminum fuselage structure. The analysis has been performed based on section properties of each design at a body station aft of the wing. The models included the fuselage skin, stiffeners, tear straps, frames, and attachment details. The interior panel configuration typical for the Boeing 757 was included in both the composite and aluminum configurations. Circumferential and longitudinal variations in the structural configuration were not assessed.

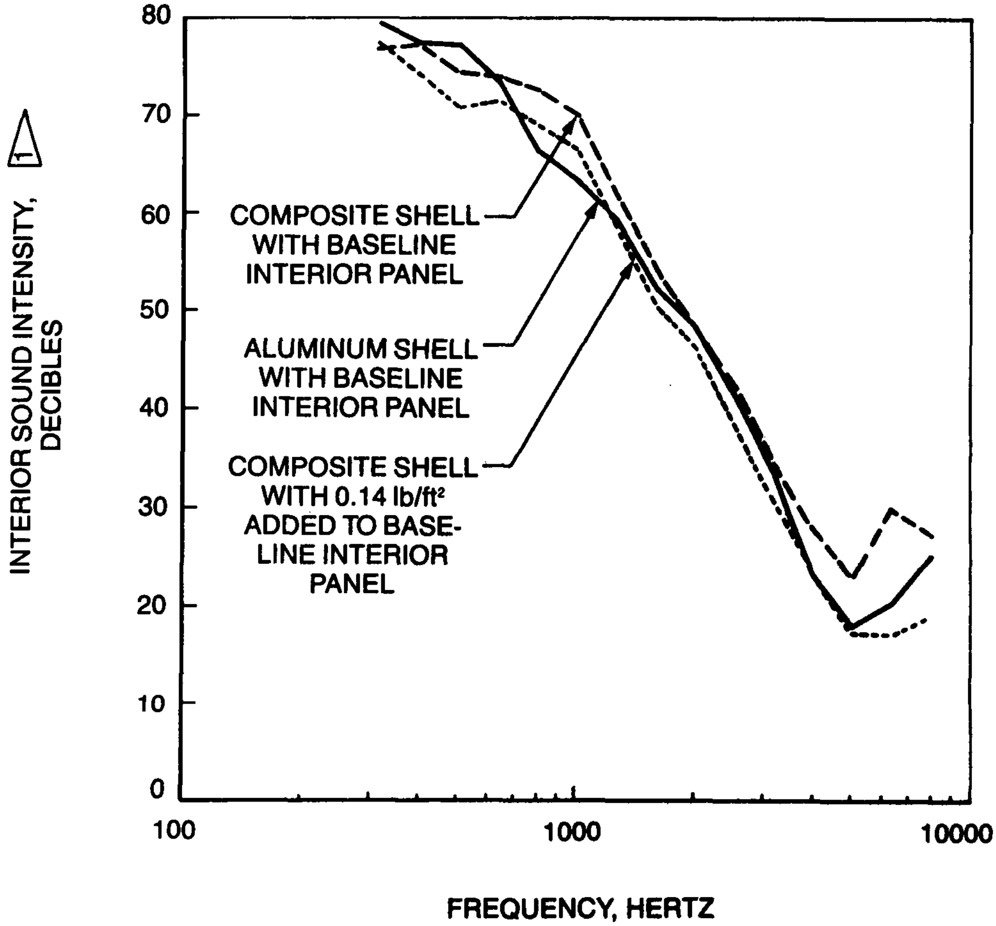
The interior noise levels of the composite and aluminum baseline configurations due to an exterior sound level of 120 decibels is shown in Figure 6.3-5. The resulting speech interference level (SIL) of the composite configuration is approximately 3.2 decibels greater than the aluminum SIL. The weight penalty associated with reducing the composite SIL to the aluminum baseline level has been assessed by adding weight to the interior panels in the composite model. By adding 0.14 lb/ft2 to the interior panel, the SIL can be reduced to 0.5 decibels below the aluminum SIL. The interior sound intensity for this configuration is shown in Figure 6.3-5. The weight penalty over the full fuselage would be approximately 280 pounds (fig. 6.3-1). Long-range solutions including acoustic damping located between structural members and attachment points will reduce the amount of sound energy transmitted through the shell at a lesser weight penalty.

6.4 MANUFACTURING

An efficient manufacturing procedure for fabrication of composite fuselage components is a basic requirement for their use since competitive cost and uniformity of quality are essential to a production program. Fabrication of composite components today, in general, is very labor dependent. Automation methods for composite part fabrication need to be developed. In addition, automated assembly methods need to be developed for composite components to be competitive with automated drilling and fastening methods presently employed for metal structure. Quality assurance needs to be maintained at a high level to insure the integrity of composite fuselage components. Automation methods for inspection also need to be developed to minimize total part cost. Methods suitable for manufacturing composite fuselage shell structures are described in Section 5.1.

6.4.1 Fabrication

The technology issue of fabrication is cost. Typically, if designs can be simplified, fabrication costs can be reduced. These simplifications almost always result in a heavier weight structure that leads to cost/weight tradeoffs. Generally, sections that are constant are less expensive to make than sections that are tapered. An example of this is shown in Figure 6.4-1. If the skin gage of the study section was not decreased and if the volume of material at the constant section was not reduced, the additional weight would be 106 pounds. This worst case weight penalty is significant, and cost effective fabrication methods need to be developed that are capable of producing tapered sections. Tapered sections could be built from flat patterns that have been laid up by automated equipment. For fuselage sections with double curvature, this method will be limited by the allowable ply distortions that will result from forcing the flat patterns into the double curvature. Filament winding methods have been considered as a possible solution; however, the ability of this method to produce changes in thickness and cross section have not been demonstrated.

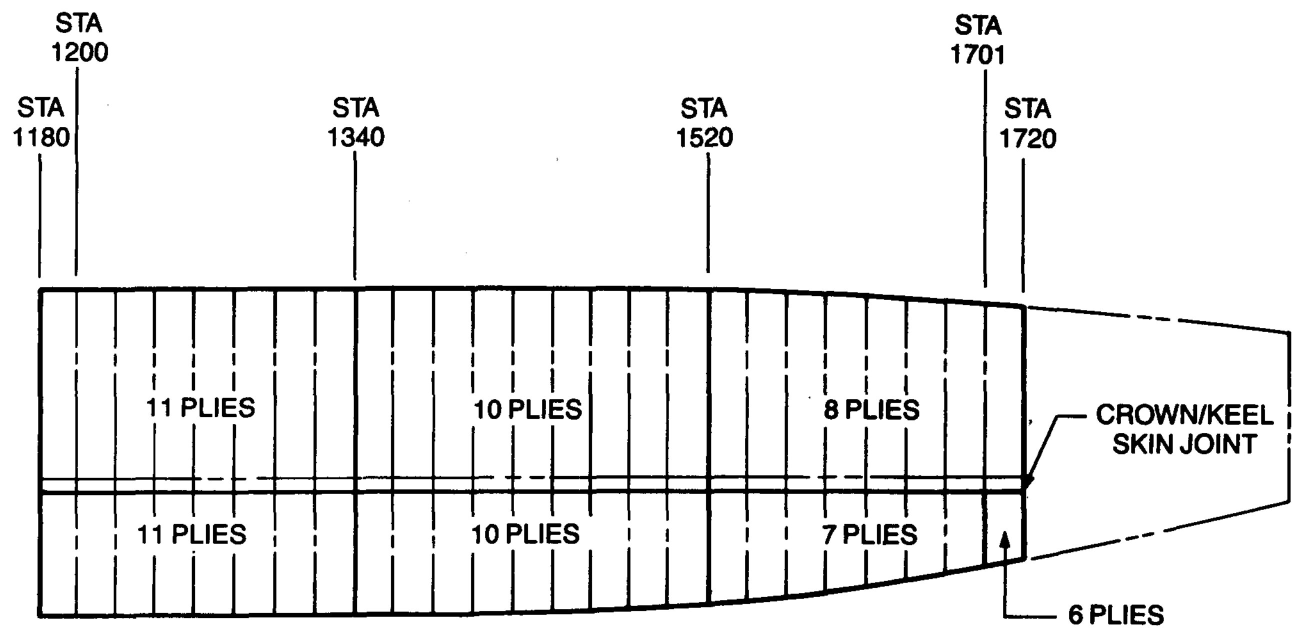


COMPOSITE CONFIGURATION BASED ON CROWN I-STIFFENER LAMINATE SKIN DESIGN AT STATION 1300

ALUMINUM CONFIGURATION BASED ON CROWN INVERTED HAT STIFFENER DESIGN AT STATION 1300

1 INTERIOR SOUND INTENSITY ESTIMATED BASED ON EXTERIOR SOUND PRESSURE LEVEL OF 120 DECIBELS

Figure 6.3-5. Interior Noise Level of Baseline Aluminum and Composite Fuselage Concepts



SKIN GAGES OF CONCEPT NO. 4, HAT STIFFENED LAMINATE DESIGN

I. REFINED FABRICATION WEIGHT OF STUDY SECTION (CONCEPT 4) WITH SKIN GAGES AS SHOWN ABOVE:

2590 lb (28% WEIGHT REDUCTION [1])

II. NONREFINED FABRICATION WEIGHT OF STUDY SECTION (CONCEPT 4) WITH SKINS FABRICATED BY MAINTAINING CONSTANT 10-PLY COUNT AND SURFACE AREA FROM CONSTANT SECTION THROUGH TO STATION 1720:

2696 lb (26% WEIGHT REDUCTION [1])

DIFFERENCE IN STUDY SECTION WEIGHT BETWEEN NONREFINED AND REFINED FABRICATION METHODS:

2696 - 2590 = 106 lb

[1] WEIGHT REDUCTION COMPARED TO ALUMINUM BASELINE STUDY SECTION WEIGHT OF 3650 POUNDS

Figure 6.4-1. Influence of Skin Fabrication Techniques on Fuselage Weight

Body frames present a challenge for fabrication. Frames are basically curved beams and contain two chords separated by a shear web. The most efficient composite beam would contain uniaxial material in the chords and cross-plied material in the web. With these basic requirements of material placement in the curved shape, it is easily realized that the automated equipment to produce body frames will be very complex and costly to develop. Body frames could be built by hand layup using sections of cross-plied fabric or tape for the web combined with uniaxial material for the chords. The modulus of the frame web would vary over the length of a frame fabricated this way, as shown in Figure 6.4-2. Extra plies will have to be added to the webs to provide splice material for the sections and also provide extra thickness to account for variations in shear modulus around the curved shape.

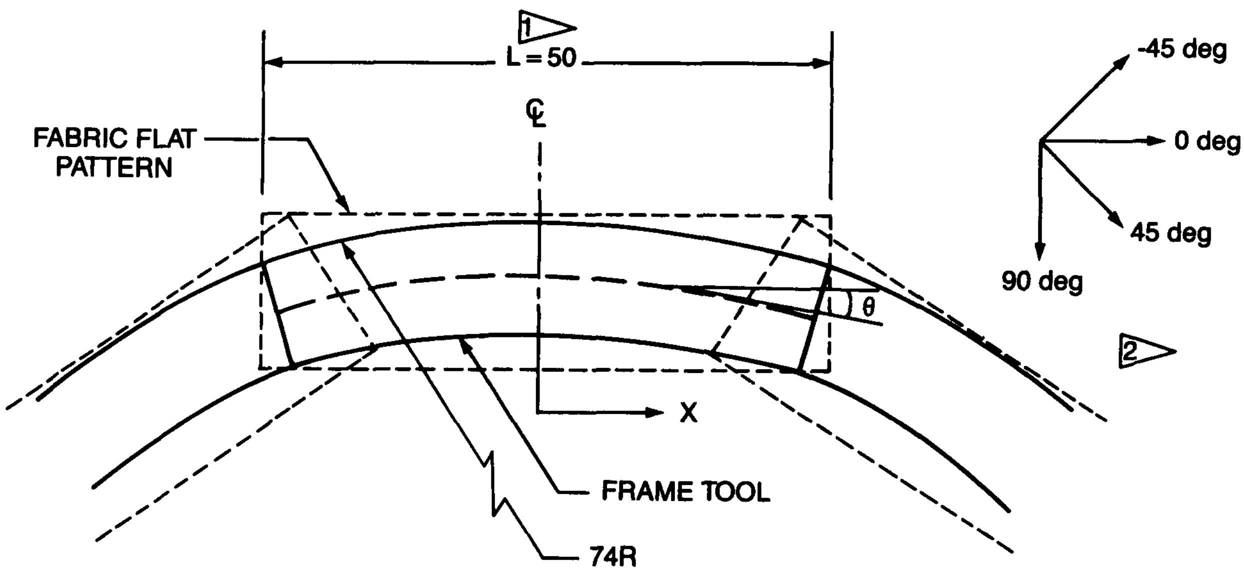
This discussion demonstrates that restrictions imposed by laminating and trimming processes can affect fuselage structural properties and weight. Trade studies need to be performed between fabrication cost and design to arrive at a cost effective design.

6.4.2 Assembly

The primary concern in composite fuselage assembly is that current assembly procedures for aluminum structure are developed for smaller sized parts and a higher number of subassemblies. The three panel assembly approach described in Section 5.1.2, and shown in Figures 5.1-11 and 5.1-12, will need to be automated to be cost effective. Since a high degree of cocuring can be done during fabrication, part size for assembly will be larger than in an aluminum fuselage shell. Techniques need to be developed for handling and positioning these larger parts. In addition, automated drilling procedures (fig. 5.1-13, for example) need to be developed.

6.4.3 Quality Control

The primary task in the area of quality control is how to provide an adequate level of inspection in a cost effective manner in a production environment. Ultrasonic through transmission and X-ray methods have been developed and have proven adequate for nondestructive inspection of composite parts. However, these methods will have to be adapted to fuselage structure where there will be large surface areas and parts with single and double curvature. The main concern is not a requirement to develop new techniques but to adapt existing methods to the new requirements. In addition, the inspection equipment will have to be automated to be cost effective.



HORIZONTAL DISTANCE OFF OF FRAME C_L X, in	θ-ANGULAR LAMINATE ORIENTATION CHANGE, deg	FRAME WEB MODULI IN RADIAL DIRECTION [3]	
		E Msi	G Msi
0	0	5.82	3.22
5	4	5.85	3.19
10	8	5.94	3.09
15	12	6.08	2.95
20	16	6.27	2.78
25	20	6.51	2.61

[1] 50-in LENGTH IS MAXIMUM FLAT PATTERN LENGTH ATTAINABLE FROM 42-in WIDE FABRIC MATERIAL CUT AT ±45deg ORIENTATION

[2] FABRIC FLAT PATTERN ORIENTATION AT X = 0 REMAINS FIXED FOR ALL POSITIONS ALONG FRAME FOR X = ±25 in

[3] BASELINE LAMINATE AT X = 0: (45, –45)/(0,90)/(45,-45)
TYPE II, 3K-70-PW FABRIC

$E_{11} = E_{22} = 9.3$ msi, $G_{12} = 0.90$ msi, $\nu_{12} = 0.06$

Figure 6.4-2. Modulus Variances in Curved Composite Frames

6.5 TECHNOLOGY ISSUE PRIORITIES

The principal research requirements and priorities within the categories of structures, materials, systems, and manufacturing are summarized in Figures 6.5-1 through 6.5-4. All technology issues need to be addressed and the priority levels are defined to serve as a guide to establishing budget and schedule priorities. The urgency for resolving technology issues is influenced by schedule requirements. If given enough time and resources, all of the technology issues can eventually be resolved. The order in which the issues are addressed, though, is influenced by the design development process. In any technology development program, the fundamental material and structural requirements must be evaluated in terms of influencing the initial design configuration. For composite fuselage structure, this includes the development of material specifications, and design of structural details such as joints, splices, attachments, and windows. In addition, analysis procedures for pressure damage containment, stability, postbuckling, and impact resistance must be developed and verified.

Technology development programs for issues that are significantly influenced by configuration should be assessed, starting after the fuselage design configuration has been outlined. These configuration-related technologies include structural requirements for impact dynamics, electromagnetic protection, acoustic transmission, and repair. In addition, the influence that temperature and moisture have on material properties will have to be incorporated into the development process. All of the technology developments should be incorporated into the final design and then substantiated during a verification test program. In order to ensure fuselage producibility, manufacturing technology must be developed parallel to the structural development.

RELATIVE PRIORITY RANKING	TECHNOLOGY ISSUE	PRINCIPAL RESEARCH REQUIREMENT PRIORITIES
1 (HIGHEST PRIORITY)	PRESSURE DAMAGE CONTAINMENT	• LARGE DAMAGE FRACTURE RESPONSE • TEAR STRAP DESIGN PROCEDURE VERIFICATION • EVALUATION OF CORRECTION FACTOR (K) FOR TEMPERATURE, MOISTURE, PRESSURE, CURVATURE
2	POSTBUCKLED STRUCTURE	• SKIN-STRINGER INTERFACE STRESS ANALYSIS AND ALLOWABLES • ULTIMATE STRENGTH DESIGN ANALYSIS FOR STRINGER-STIFFENED PANELS
3	IMPACT DYNAMICS	• SUITABILITY OF FAA LOAD FACTOR REQUIREMENTS • DYNAMIC ANALYSIS OF FUSELAGE RESPONSE • TEST VERIFICATION
4	BOLTED JOINTS	• BEARING-BYPASS ALLOWABLES • BIAXIAL LOADING • SPLICE DESIGN
5	CUTOUTS	• REINFORCEMENT DESIGN • BIAXIAL AND INTERLAMINAR STRENGTH ALLOWABLES
6	REPAIR	• DESIGN-REPAIR COST TRADE STUDIES • REPAIR ADEQUACY VERIFICATION

Figure 6.5-1. Structures Technology Research Priorities

RELATIVE PRIORITY RANKING	TECHNOLOGY ISSUE	PRINCIPAL RESEARCH REQUIREMENT PRIORITIES
1 (HIGHEST PRIORITY)	DESIGN STRAIN LEVELS AND IMPACT DAMAGE	• IMPACT DAMAGE LEVEL REQUIREMENTS • MATERIAL SELECTION • DESIGN CONFIGURATION
2	FLAMMABILITY AND FIRE PROTECTION	• ASSESS EMERGING REQUIREMENTS • VERIFY MATERIAL SUITABILITY

Figure 6.5-2. Materials Technology Research Priorities

RELATIVE PRIORITY RANKING	TECHNOLOGY ISSUE	PRINCIPAL RESEARCH REQUIREMENT PRIORITIES
1 (HIGHEST PRIORITY)	ACOUSTIC TRANSMISSION	• NOISE ATTENUATION ANALYSIS • INSULATION MASS REQUIREMENTS • DESIGN CONFIGURATION • NOISE LEVEL VERIFICATION
2	ELECTROMAGNETIC EFFECTS	DETERMINE AND VERIFY DESIGN APPROACHES FOR ADDRESSING REQUIREMENTS OF: • ELECTRICAL RETURNS • ELECTROMAGNETIC SHIELDING • PERSONNEL PROTECTION
3	LIGHTNING PROTECTION	• DESIGN DEVELOPMENT • DESIGN VERIFICATION

Figure 6.5-3. Systems Technology Research Priorities

RELATIVE PRIORITY RANKING	TECHNOLOGY ISSUE	PRINCIPAL RESEARCH REQUIREMENT PRIORITIES
1 (HIGHEST PRIORITY)	FABRICATION	• COST • AUTOMATION • COST REDUCTION • DEVELOP METHODS FOR COMPLEX SHAPES • COCURING OF COMPLEX ASSEMBLIES
2	ASSEMBLY	• COST • AUTOMATION • COMPLEX SHAPE ASSEMBLY • FASTENING TECHNIQUES
3	QUALITY ASSURANCE	• COST • AUTOMATION • ACCURACY

Figure 6.5-4. Manufacturing Technology Research Priorities

7.0 DEVELOPMENT PROGRAM

Eleven development program elements, which will provide the technology data base necessary to commit composites to fuselage structure, were defined. The cost of these program elements was estimated and schedules were defined. Five program options, containing combinations of these program elements, are defined and discussed and a Boeing selected option plan is presented.

7.1 DEVELOPMENTAL PROGRAM ELEMENTS

A total of eleven developmental program elements with their objectives are presented in Figure 7.1-1. The detailed test plans for program elements I, IV, V, VI, VII, and VIII are presented in Appendix A. A summary of each of the program elements is presented as follows.

Coupons and Subcomponents (Element I) — This program element provides basic material property data and strength and damage tolerance of basic fuselage panels and associated components. Temperature and moisture effects are obtained for all test configurations. The details of this program element are summarized as follows:

- 1526 basic material property coupons (400 material allowable coupons, 72 material fracture coupons, 1000 mechanical fastened joint coupons, 54 tension fittings)
- 50 fracture panels
- 108 crippling elements
- 54 frame to skin out-of-plane pressure loaded test components
- 36 frame bending elements
- 36 frame shear tie elements
- 30 shear-tension-compression-pressure combined load panels
- 27 window frame panels
- 36 combined load splice details.

Systems (Element II) — This program element establishes the adequacy of current technology and advances the state-of-the-art where possible, to provide protection to passengers and electrical/electronic equipment against direct lightning strike and induced transients. This program element includes tasks to determine noise attenuation effects and provides equivalent noise levels, as presently occur in aluminum airplanes, with a minimum of added weight. This element contains the following:

- 10 lightning strike panels
- Fiber optic development components
- Electrical/electronic shielding component parts
- 12 noise attenuation test panels
- 30-foot-long full-scale composite fuselage section

PROGRAM ELEMENT	DESCRIPTION	OBJECTIVE(S)
I	COUPONS AND SUBCOMPONENTS	• DETERMINE MATERIAL PROPERTIES AND ALLOWABLES • EVALUATE LOCAL STRUCTURAL DETAILS • ANALYSIS VERIFICATION

II	SYSTEMS TESTS	• VERIFY ADEQUACY OF SYSTEMS PROTECTION
		  • DIRECT LIGHTNING
		  • ELECTRICAL/ELECTRONICS
		• VERIFY ADEQUACY OF NOISE ATTENUATION

III	IMPACT DYNAMICS	• VERIFY PASSENGER SAFETY UNDER CONTROLLED IMPACT CONDITIONS
IV	ENVIRONMENTAL TESTS	• ESTABLISH STRENGTH REDUCTIONS DUE TO MOISTURE AND TEMPERATURE • ESTABLISH FAA CERTIFICATION BASE
V	REPAIR	• EVALUATE REPAIR METHODS
VI	QUARTER PANEL TESTS	• EVALUATE FULL-SCALE REPRESENTATIVE FUSELAGE PANELS WITH COMBINED LOADINGS
VII	FULL-SCALE AFTBODY TEST	• VERIFY STRUCTURAL CAPABILITY OF FULL-SCALE FUSELAGE AFTBODY SECTION
VIII	FULL-SCALE CENTER SECTION TEST	• VERIFY STRUCTURAL CAPABILITY OF FULL-SCALE FUSELAGE CENTER SECTION
IX	MANUFACTURING TECHNOLOGY—SHELL COMPONENTS	• DEVELOP AUTOMATION METHODS FOR SHELL COMPONENTS • VERIFY METHODS • VERIFY COST REDUCTIONS
X	MANUFACTURING TECHNOLOGY—NON-SHELL COMPONENTS	• DEVELOP AUTOMATION METHODS FOR NON-SHELL COMPONENTS • VERIFY METHODS • VERIFY COST REDUCTIONS
XI	FLIGHT TEST	• PERFORM FLIGHT TEST • OBTAIN 1-YEAR SERVICE EXPERIENCE

Figure 7.1-1. Program Elements

The program plan for the full-scale fuselage section will be to conduct system shielding tests and noise attenuation tests. At the completion of this series of tests, the full-scale fuselage section could then be used to verify passenger safety under controlled impact conditions.

Impact Dynamics (Element III) — This program element consists of a number of coupon, component, and subcomponent tests to determine the best structural configurations for frames, floor beams, and skin panels to absorb energy during emergency landing conditions. Based on these test results, combined with analysis, promising structural components can be designed and included in a full-scale fuselage section which would be subjected to a controlled impact test to establish equivalency to an aluminum fuselage section.

Environmental Coupons and Subcomponents (Element IV) — This program element provides the basis for the approach to be used to obtain FAA certification of a graphite composite fuselage. The elements in this program are:

- 125 basic material coupons
- 30 shear panel subcomponents.

The program establishes basic strength, damage growth, and residual strength of the laminate material and shear panel subcomponents before and after simulated real-time temperature, moisture, and load. The primary objective of this test program is to demonstrate that full-scale static, durability, and damage tolerance tests conducted under room temperature ambient conditions will provide the substantiating evidence needed to fulfill the FAA requirements.

Repair (Element V) — This program element establishes the adequacy of repair procedures at temperature and moisture extremes for panels and components. The details of this program element are summarized as follows:

- 12 shear-tension-compression-pressure combined load panels
- 6 window frame panels

Quarter Panel Tests (Element VI) — This program element provides verification of the design of major sized fuselage panels for ultimate strength and damage tolerance. The test parts included in this program are:

- 2 - 100-in by 180-in pressure-shear-tension-compression damage containment panels
- 2 - 60-in by 100-in compression-shear damage containment panels
- 2 - 60-in by 100-in window frame ultimate strength panels
- 2 - 80-in by 120-in keel beam redistribution ultimate strength and damage tolerance panels

Full-Scale Aftbody (Element VII) — This program element provides ultimate strength, durability, damage tolerance and residual strength for a complete 45-foot long aftbody fuselage section. The significant details that are included in this test article are the keel beam, the aft wheel well bulkhead, and the aft pressure bulkhead. This test article contains a floor beam loading system to simulate passenger and cargo inertia loads.

Full-Scale Center Section (Element VIII) — This program element provides ultimate strength, durability, damage tolerance and residual strength for a 50-foot long fuselage center section. The test article includes an aftbody section, a forward body section, center section, and a left and right hand stub wing. The significant details that are included in this test article are front and rear spar bulkheads, aft wheel well bulkhead, keel beam, and door cutouts. This test article contains a floor beam loading system to simulate passenger and cargo inertia loads.

Manufacturing Technology-Fuselage Shell Structure (Element IX) — This program element covers the development and demonstration of generic fabrication methods for composite fuselage structure. Automated processes, using state-of- the-art technology, are demonstrated for fabrication, assembly, and inspection of the basic shell structure. The goal is to reduce manufacturing costs for a composite fuselage shell by 10%, compared to the equivalent aluminum structure.

Manufacturing Technology-Nonshell Structural Elements (Element X) — This program element addresses the manufacturing of structural components that are not part of the basic fuselage shell. The components include major bulkheads, floor beams, window and door frames, and wing-to-body and empennage-to-body joints. Materials and processes such as thermoplastics, automated fabrication and assembly, and associated quality control technologies would be developed.

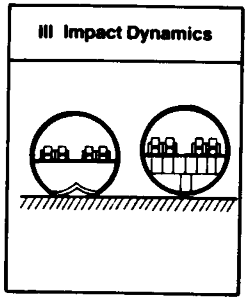
Flight Test (Element XI) — This program element includes the fabrication and installation of a 20-foot long section of fuselage to be installed in a 757 airplane. This airplane would be leased by Boeing and the composite section would be installed. The airplane is put into airline service for one year to obtain service experience. At the end of this service period, the composite section is removed, the metal section reinstalled, and the airplane returned to revenue service.

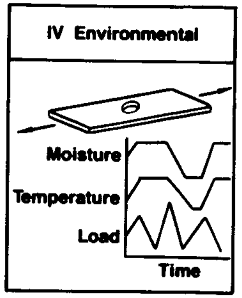
7.2 PROGRAM OPTIONS AND SELECTED PROGRAM

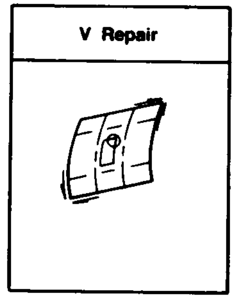
A total of five program options have been defined by combining selected program elements. These program options, including estimated labor-years of effort, are presented in Figure 7.2.1. The estimates include engineering, fabrication, assembly, test hours, and material costs converted to labor- years. Each program option contains engineering design development hours to integrate the technology solutions into the final design.

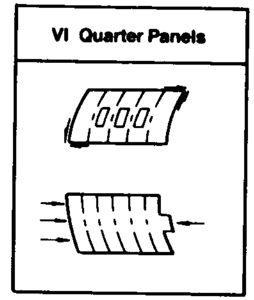
The labor estimate for Element II, Systems, contains engineering design, fabrication, and assembly hours and materials for a 30-foot long full-scale fuselage section with windows and no doors. The estimate for Element III, Impact Dynamics, does not contain labor for a full-scale section. The systems tests performed on the full-scale section would be nondestructive; thus, the same fuselage section could be used for the impact dynamics tests. In addition, the labor estimates developed for the Impact Dynamics program do not include testing for the full-scale section impact test. It has been assumed that this phase of the program would be performed by NASA personnel at the Impact Test Site at NASA Langley.

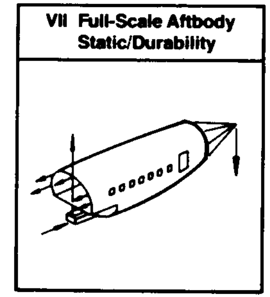
Option 1 (fig. 7.2-1) contains elements I, II, III, IV, V, VI, and IX. These program elements are considered as a minimum base to establish fuselage technology readiness. Option 2 contains the same elements as in Option 1 with the quarter panel tests (Element VI) replaced by the full scale aftbody test (Element VII). Option 3 contains the same elements as Option 2 with the quarter panel program (Element VI) and the nonshell components manufacturing technology program (Element X) added. Option 4 contains the same elements as Option 3 with the full scale aftbody test (Element VII) replaced by the full- scale center section test (Element VIII). Option 5 contains the same elements as Option 4 with a flight test program (Element XI) added.

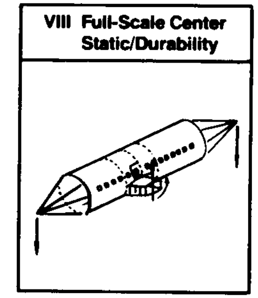


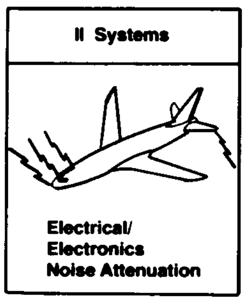




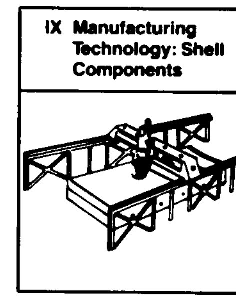


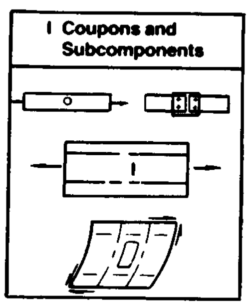






	PROGRAM ELEMENTS	PROGRAM OPTION NUMBER				
		1	2	3	4	5
1	COUPONS AND SUBCOMPONENTS	x	x	x	x	x
II	SYSTEMS	x	x	x	x	x
III	IMPACT DYNAMICS	x	x	x	x	x
IV	ENVIRONMENTAL	x	x	x	x	x
V	REPAIR	x	x	x	x	x
VI	QUARTER PANELS	x		x	x	x
VII	FULL-SCALE AFTBODY STATIC/DURABILITY		x	x		
VIII	FULL-SCALE CENTER SECTION STATIC DURABILITY				x	x
IX	MANUFACTURING TECHNOLOGY: SHELL COMPONENTS	x	x	x	x	x
X	MANUFACTURING TECHNOLOGY: NON-SHELL COMPONENTS			x	x	x
XI	FLIGHT TEST					x
	TOTAL PROGRAM COST LABOR—YEARS	600	750	1000	1250	2000-2500





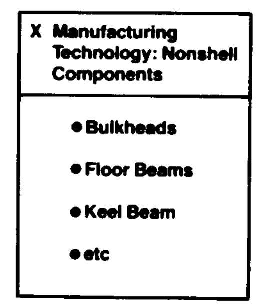


Figure 7.2-1. Program Options Summary

The five options (Figure 7.2-1) were evaluated for technology readiness risk. The results of this risk assessment are shown in Figure 7.2-2. A definition of the requirements are as follows:

1. Panel design verification: This requirement is associated with committing a fuselage design to a full-scale structural test before performing verification tests of large fuselage panels. The quarter panel program has been specifically defined to perform this function.

2. Major load input details: This requirement is associated with verifying the design of major load introduction components such as wing to body attachment, body mounted landing gear beam attachment, and keel beam.

3. Full length section fabrication and assembly: This requirement is associated with verifying that tooling, fabrication, and assembly techniques will apply to complete fuselage sections.

4. FAA requirements for certification: This requirement is associated with the certainty of obtaining FAA certification.

5. Flight test and service experience: This requirement is associated with whether or not future production commitments would be made without first having performed a flight test and obtaining service experience.

REQUIREMENTS	[1] CONFIDENCE WEIGHTING FACTOR	OPTION 1	OPTION 2	OPTION 3	OPTION 4	OPTION 5
PANEL DESIGN VERIFICATION	20	20	5	20	20	20
CONCENTRATED LOAD INTRODUCTION DETAILS	20	5	15	15	20	20
FULL-LENGTH SECTION FABRICATION AND ASSEMBLY	20	5	15	15	20	20
FAA REQUIREMENTS FOR CERTIFICATION	30	10	15	25	30	30
FLIGHT TEST/SERVICE	10	0	0	0	0	10
TOTAL CONFIDENCE FACTOR (%)	100	40	50	75	90	100

[1] CONFIDENCE WEIGHTING FACTOR - VALUE IS EARNED WHEN ALL REQUIREMENTS HAVE BEEN ACHIEVED

Figure 7.2-2. Program Option Risk Assessment

As noted in Figure 7.2-2, the requirements have been assigned different confidence weighting factors depending upon the capability of achieving each requirement. Performing a flight test and obtaining service experience has been assigned the lowest weighting factor due to the large costs involved and short service planned. It is considered that one year of service would not provide representative data. In addition, performing a flight test to determine if the composite fuselage section changes the aircraft handling characteristics is not considered necessary since the stiffness of the composite section will be similar to the existing aluminum section.

Achieving FAA certification for a composite fuselage section prior to a production commitment has been assigned the highest weighting factor.

The resource requirements and risk assessments for each of the five program options are shown in Figure 7.2-3. The program length for Option 1 would be five years, the program length for Options 2, 3, and 4 would be eight years and Option 5 would be nine years including the one year of service experience. Based on a review of the program costs, schedules, and risk assessments, Boeing has selected Option 3 as the preferred technology development program.

Option 3 provides 75% of the requirements, which is considered an acceptable risk level. The detailed schedule for the Option 3 program is shown in Figure 1.0-6. Option 4 was not selected due to the 25% additional program cost. Option 5 was not selected since the program cost outweighed the additional benefits. Options 1 and 2 were not selected as they presented too high a risk.

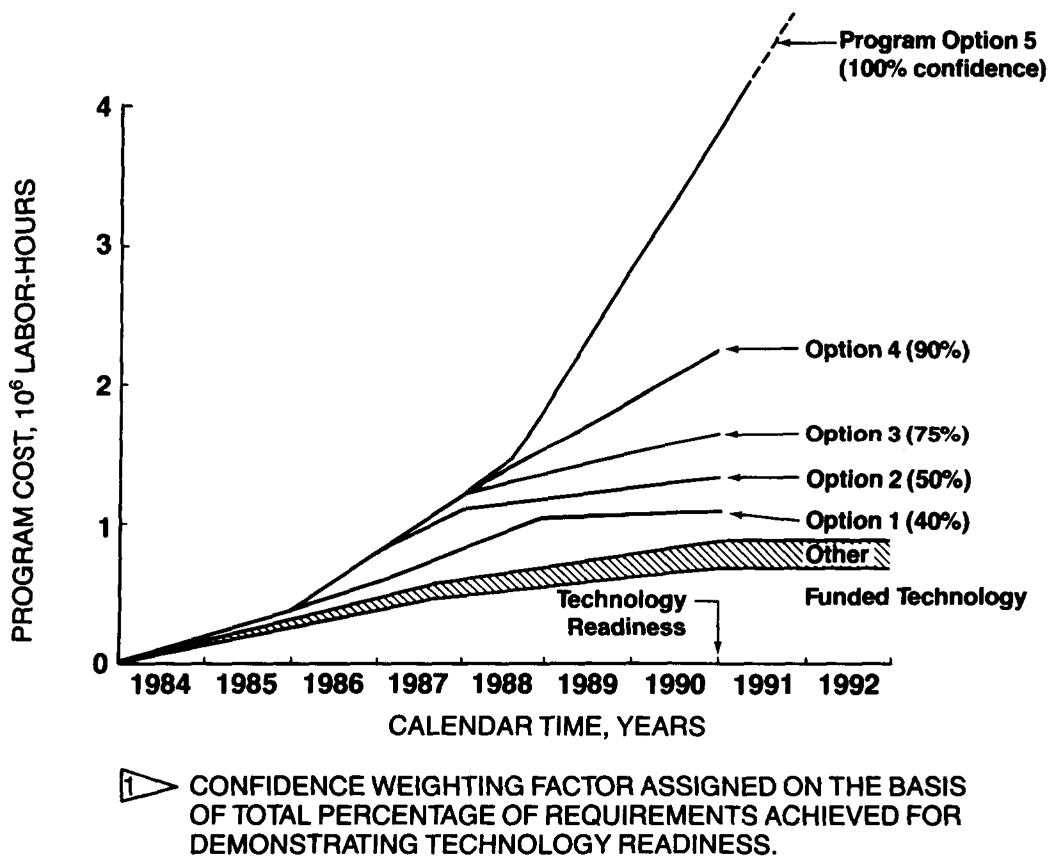


Figure 7.2-3. Resource Requirements and Risk Assessment for Each of Five Program Options

8.0 PROGRAM CONCLUSIONS AND RECOMMENDATIONS

This study program has been performed to define and plan a development program directed towards achieving technology readiness to support the introduction of advanced composite material in fuselage structure of future commercial and military transport aircraft. Composite fuselage design concepts have been developed and relative costs and weights have been estimated. Two design concepts, I-section stiffened laminate skin panels and honeycomb stabilized skin panels, were selected to be carried forward into the developmental program. Major technology issues have been defined and their significance in relation to the overall technology development program has been discussed. These technology issues are defined as:

- Materials
  - Flammability and fire protection
  - Design strain levels
  - Impact damage
- Structures
  - Pressure damage containment
  - Stability and post buckling
  - Joints, splices, and attachments
  - Cutouts
  - Impact dynamics
  - Repair
- Systems
  - Lightning protection
  - Electromagnetic effects
  - Acoustic transmission
- Manufacturing
  - Fabrication
  - Assembly
  - Quality control

Technology development program elements have been defined and cost estimates have been obtained. Five program options have been defined and Option 3 has been selected as Boeing's preferred plan. The selected option contains programs that address all of the aforementioned technology issues and includes a static and durability test of a full-scale fuselage aftbody section.

The selected option has been scheduled as an eight-year development program leading to technology readiness in the early 1990s.

The proposed fuselage program is a logical and timely follow-on to the current NASA, Air Force, and industry graphite-epoxy development and production programs. A 20-30% weight reduction in participating fuselage structure compared with current aluminum fuselages is attainable, and would contribute significantly to the NASA/ACEE program goal of significantly improving fuel efficiency and range capability of commercial and military transports. The cost to develop advanced composties for fuselage application is acceptable when balanced against the potential fuel savings and manufacturing economics.

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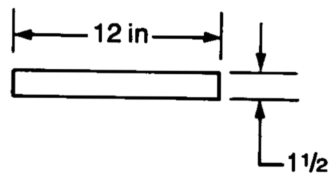
APPENDIX A PROGRAM ELEMENT TEST PLANS

CONTENTS TO APPENDIX A

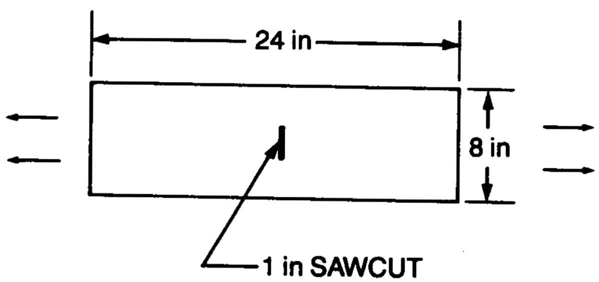
CONTENTS TO APPENDIX A (Continued)

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

1	MATERIAL ALLOWABLE COUPONS
	
	• 400 COUPONS
	• THICKNESS OF COUPONS VARY BETWEEN 8 AND 20 PLIES
	• SPECIMEN VARIATIONS INCLUDE:
	  • UNDAMAGED, WITHOUT HOLE
	  • OPEN HOLE IN CENTER
	  • IMPACT DAMAGE
	  • DISCONTINUOUS LAMINATES
	  • PLY ORIENTATION
		• STATIC TEST COUPONS TO FAILURE
		• TEST 200 COUPONS IN TENSION AND 200 COUPONS IN COMPRESSION
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • −65° DRY
		  • 160°F WET

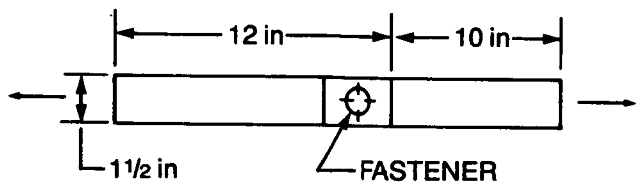
Program Element I, Material Allowable Coupons

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

2	MATERIAL FRACTURE COUPONS
	
	
	• 72 COUPONS • 36 TENSION
	• 36 COMPRESSION
	• 8 AND 20 PLIES THICK
	• 4 LAMINATE ORIENTATIONS
		• STATIC TEST TO FAILURE
		• STABILIZATION PLATES WILL BE USED DURING TEST
		• RECORD DATA FOR 6 BACK-TO-BACK GAGES
		• RECORD CRACK OPENING DISPLACEMENT
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160° WET
		  • −65°F DRY

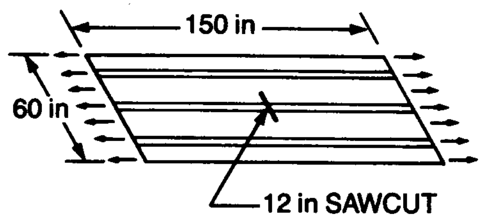
Program Element I, Material Fracture Coupons

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

3	MECHANICALLY FASTENED JOINT ALLOWABLES
	
	• THICKNESS VARIES BETWEEN 0.040 AND 0.125 INCH
	• FASTENER SIZES INCLUDED ARE 5/32, 3/16, 1/4 AND 5/16 INCH
	• 1000 SPECIMENS
	• SPECIMEN VARIATIONS INCLUDE
	  • COUNTERSUNK FASTENERS
	  • PROTRUDING HEAD FASTENERS
	  • SINGLE SHEAR
	  • DOUBLE SHEAR
	  • LOAD DIRECTION
		• STATIC TEST COUPONS ALL COUPONS TO FAILURE IN TENSION
		• RECORD LOAD VS DEFLECTION
		• TEST ENVIRONMENT:
		  • 70°F DRY
		  • 160°F WET
		  • −65° DRY

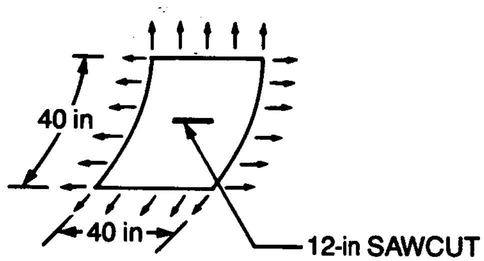
Program Element I, Mechanically Fastened Joint Allowables

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

4	FLAT LAMINATE FRACTURE PANEL
	
	
	• 25 PANELS
	  • 15 PANELS 30 in x 120 in — 10-in SPACING
	  • 10 PANELS 60 in x 150 in — 20-in SPACING
	  • 8 TO 20 PLIES THICK
	• TEAR STRAP STIFFENING FROM 0 TO 75%
		• STATIC TEST ALL PANELS TO FAILURE IN TENSION
		• CRACK STABILIZATION PLATES WILL BE PROVIDED
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • −60°F DRY
		• RECORD STRAINS FOR 27 AXIAL GAGES AND 6 ROSETTE GAGES
		• RECORD CRACK OPENING DISPLACEMENT
		• HIGH SPEED MOVIES WILL BE TAKEN OF FRACTURE AREA FOR 30% OF THE PANELS
		• 12 BASIC MATERIAL COUPONS WILL BE INSTRUMENTED AND TESTED FOR EACH LAMINATE LAYUP

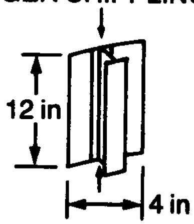
Program Element I, Flat Laminate Fracture Panels

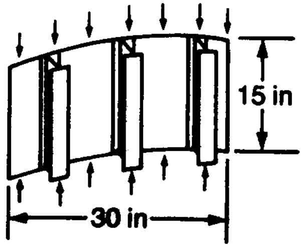
TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

5	CURVED LAMINATE FRACTURE PANEL
	
	• 24 PANELS
	• TWO LAMINATE LAYUPS
	• NO TEAR STRAPS
	• THICKNESS OF PANELS
	  • 12 PANELS 8 PLIES
	  • 12 PANELS 10 PLIES
	• 74-in RADIUS
		• 12 PANELS WITH SAWCUT SEALED WITH RUBBER MEMBRANE — PRESSURE APPLIED UNTIL FAILURE
		• 12 PANELS PRESSURE LOADED CUT WITH GUILLOTINE
		• ALL PANEL EDGES RESTRAINED
		• STRAINS RECORDED FOR 8 AXIAL AND 8 ROSETTE GAGES
		• CRACK OPENING DISPLACEMENTS WILL BE RECORDED
		• 12 BASIC MATERIAL COUPONS WILL BE INSTRUMENTED AND TESTED FOR EACH LAYUP

Program Element I, Curved Laminate Fracture Panels

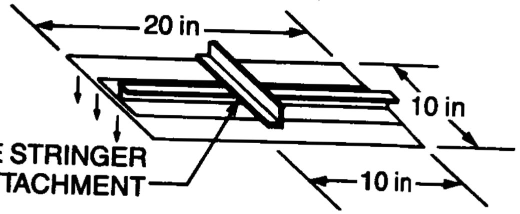
TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

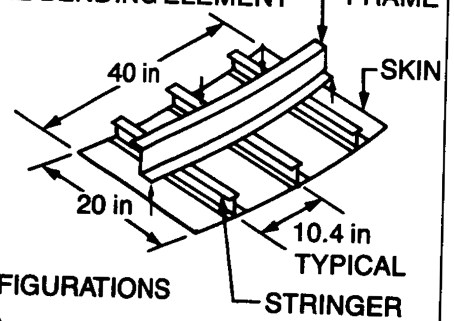
6	SINGLE STRINGER CRIPPLING ELEMENT
	
	
	• 54 TEST ELEMENTS
	• 9 DIFFERENT CONFIGURATIONS (THICKNESSES AND PLY ORIENTATION)
		• STATIC LOAD ELEMENTS TO FAILURE IN COMPRESSION
		• RECORD STRAINS FROM 8 AXIAL GAGES
		• RECORD LOAD VS DEFLECTION
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160° WET
		  • −65°F DRY

7	CURVED SKIN-STRINGER COMPRESSION PANELS
	
	
	• 54 PANELS
	• 74-in RADIUS
	• 9 DIFFERENT CONFIGURATIONS
		• STATIC LOAD PANELS TO FAILURE IN COMPRESSION
		• NONLOADED EDGE IS UNSUPPORTED
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY
		• RECORD STRAINS FROM 18 AXIAL STRAIN GAGES
		• RECORD LOAD VS DEFLECTIONS

Program Element I, Single Stringer Crippling Elements and Curved Skin-Stringer Compression Panels

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

8	PRESSURE PILLOWING PANEL (SKIN PANEL-FRAME ATTACHMENT)
	
	
	• 54 PANELS
	• 6 DIFFERENT CONFIGURATIONS
		• APPLY TENSION LOAD TO FRAME SEGMENT UNTIL FAILURE
		• NARROW EDGE OF PANEL RESTRAINED
		• LOAD VS DEFLECTION RECORDED AT TWO POINTS ON FRAME ATTACHMENT ANGLE
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY

9	FRAME BENDING ELEMENT
	
	
	• 36 PANELS
	• 4 AREA CONFIGURATIONS
	• 74-in RADIUS
		• STATIC LOAD TO FAILURE IN 4 POINT BENDING
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY
		• RECORD STRAINS FOR 18 AXIAL GAGES AND 2 ROSETTE GAGES
		• RECORD DEFLECTIONS

Program Element I, Pressure Pillowing Panels and Frame Bending Elements

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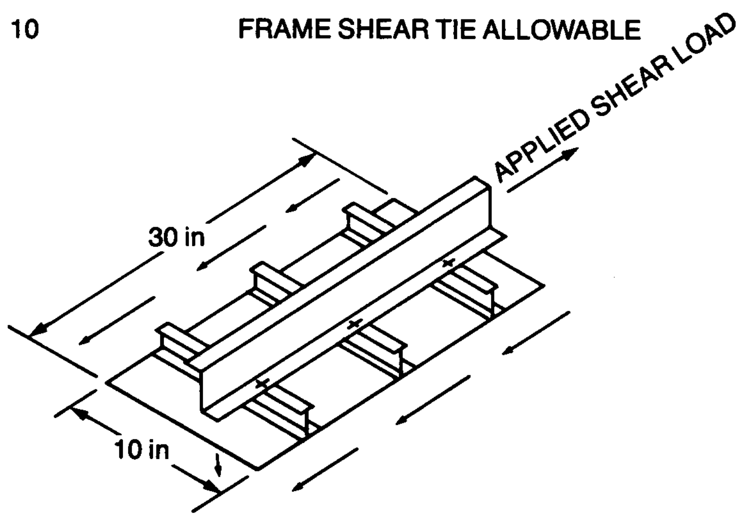
<tr><td>10</td><td>FRAME SHEAR TIE ALLOWABLE


• 36 PANELS
• 4 AREA CONFIGURATIONS
• 3 STRINGERS
• SKIN THICKNESS 8 TO 16 PLIES
• FLAT SKIN PANELS</td><td>• STATIC LOAD TO FAILURE
• ENDS OF FRAMES SIMPLY SUPPORTED
• RECORD STRAINS FOR 8 AXIAL AND 8 ROSETTE GAGES
• TEST ENVIRONMENT
  • 70°F DRY
  • 160°F WET
  • −65°F DRY</td></tr>
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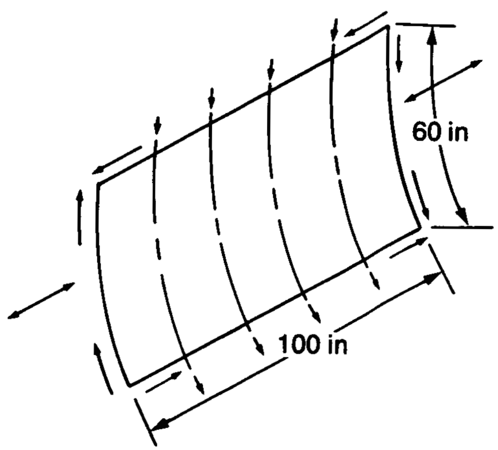
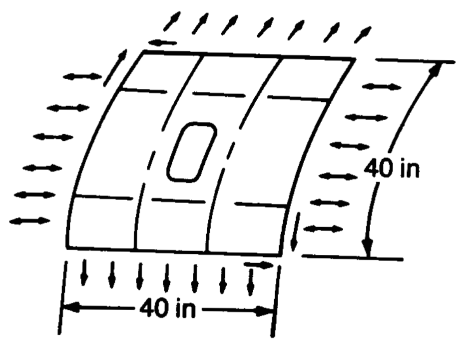
Program Element I, Frame Shear Tie

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

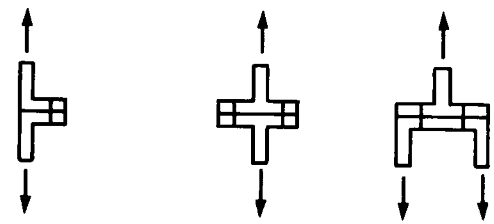
11	SHEAR AND COMPRESSION-TENSION PANELS
	
	
	• 30 PANELS
	• 74-in RADIUS
	• 4 AREA CONFIGURATIONS
	• 4 FRAMES, 5 STRINGERS
		• PANELS TESTED IN COMBINATIONS OF SHEAR, TENSION, OR COMPRESSION AND PRESSURE
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY
		• RECORD DATA FROM 20 AXIAL GAGES, 20 ROSETTE GAGES AND 10 EDIs
		• CONDUCT 4 STRAIN SURVEYS WITH VARYING LOAD COMBINATIONS PER PANEL
		• STATIC LOAD EACH PANEL TO FAILURE WITH SPECIFIED LOAD COMBINATIONS

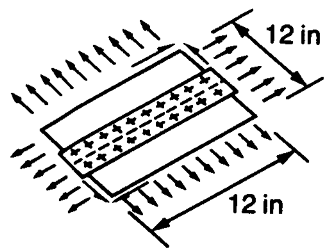
Program Element I, Shear and Compression-Tension Panels

<table>
<tr><th>TEST NUMBER</th><th>TEST ARTICLE DESCRIPTION</th><th>TEST PLAN</th></tr>
<tr><td>12</td><td>WINDOW FRAME REINFORCEMENT PANEL

• 27 PANELS
• 74-in RADIUS
• 3 AREA CONFIGURATIONS
• 1 WINDOW FRAME, 2 BODY FRAMES</td><td>• PANELS TESTED IN COMBINATIONS OF SHEAR, TENSION, OR COMPRESSION AND PRESSURE
• TEST ENVIRONMENT
  • 70°F DRY
  • 160°F WET
  • −65°F DRY
• RECORD DATA FROM 20 AXIAL GAGES, 20 ROSETTE STRAIN GAGES AND 2 DEFLECTION GAGES
• CONDUCT 4 STRAIN SURVEYS WITH VARYING LOAD COMBINATIONS PER PANEL
• STATIC LOAD EACH PANEL TO FAILURE WITH SPECIFIED LOAD COMBINATIONS</td></tr>
</table>

Program Element I, Window Frame Reinforcement Panels

TEST NUMBER	TEST ARTICLE DESCRIPTION	TEST PLAN

13	TENSION FITTINGS
	
	• 54 TEST SPECIMENS
	  • 18 BACK-TO-BACK ANGLES
	  • 18 BACK-TO-BACK TEES
	  • 18 TEE-ANGLE FITTINGS
	• THICKNESSES BETWEEN 20 AND 40 PLIES
		• STATIC LOAD TO FAILURE
		• RECORD LOAD VS DEFLECTION
		• STRAIN GAGES NOT REQUIRED
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY

14	LONGITUDINAL SKIN SPLICE
	
	
	• 36 PANELS
	• 4 LAMINATE LAYUPS
	• THICKNESS 10 TO 20 PLIES
	• FLAT PANELS — NO CURVATURE
		• TEST PANELS WITH VARYING COMBINATIONS OF BIAXIAL LOADS AND SHEAR
		• STATIC LOAD EACH PANEL TO FAILURE
		• STRAIN GAGES NOT REQUIRED
		• TEST ENVIRONMENT
		  • 70°F DRY
		  • 160°F WET
		  • −65°F DRY

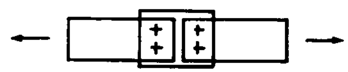
Program Element I, Tension Fittings and Longtudinal Skin Splice

TEST ARTICLE DESCRIPTION

TEST SPECIMEN DESCRIPTION (TESTS 1 THROUGH 5)

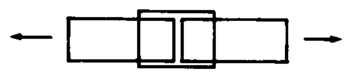
- FOUR SPECIMEN CONFIGURATIONS

CONFIGURATION 1,
CIRCULAR HOLE



CONFIGURATION 3,
MECHANICALLY
FASTENED JOINT

CONFIGURATION 2,
CUT HOLE



CONFIGURATION 4,
BONDED JOINT

- SPECIMEN SIZE 4 in BY 12 in
- TWO LAMINATE LAYUPS EACH CONFIGURATION
- 24 SPECIMENS TOTAL EACH TEST

TEST PLAN

TEST 1 — STATIC TEST

- STATIC LOAD SPECIMENS TO FAILURE
- SPECIMENS ARE NEW AND HAVE NOT BEEN CYCLED UNDER LOAD OR ENVIRONMENT
- TEST ENVIRONMENT — 70°F DRY
- RECORD LOAD VS DEFLECTION
- REPEAT FOR 24 SPECIMENS

TEST 2 — CYCLIC LOAD TEST

- CYCLE EACH SPECIMEN FOR 250,000 FLIGHTS
- TEST SPECIMENS AT CONSTANT 70°F DRY
- INSPECT FOR FLAW GROWTH EVERY 5000 FLIGHTS
- STATIC TEST EACH SPECIMEN TO FAILURE AFTER 250,000 FLIGHTS
- RECORD LOAD VS DEFLECTION
- REPEAT FOR 24 SPECIMENS

Program Element IV, Cyclic Coupon Tests

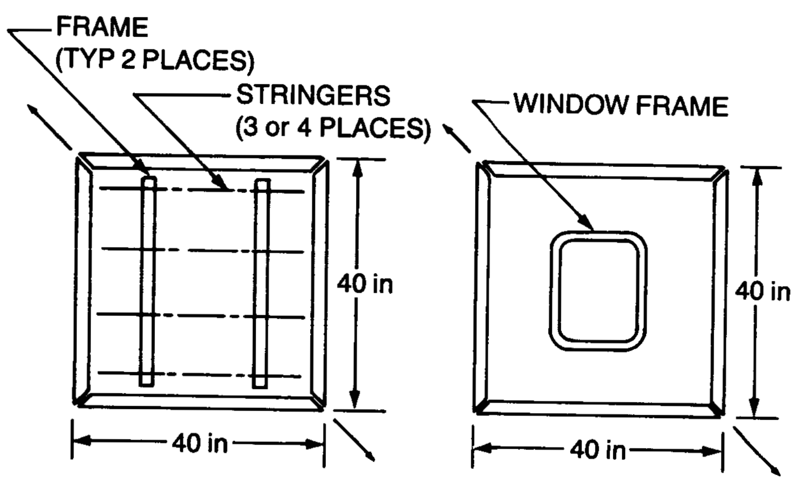
TEST ARTICLE DESCRIPTION	TEST PLAN
TEST SPECIMEN DESCRIPTION (TESTS 1 THROUGH 5)	TEST 3 — CYCLIC LOAD-ENVIRONMENT TEST • CYCLE EACH SPECIMEN FOR 250,000 FLIGHTS IN ENVIRONMENT • LOAD CYCLES (MAGNITUDE AND FREQUENCY) ARE THE SAME AS IN TEST 2 • ENVIRONMENT (TEMPERATURE AND HUMIDITY) WILL RANGE BETWEEN – 75°F (DRY) AND + 160°F (WET) • INSPECT FOR FLAW GROWTH EVERY 5000 FLIGHTS • STATIC TEST EACH SPECIMEN TO FAILURE AFTER 250,000 FLIGHTS • CONDUCT STATIC TEST AT 70°F DRY • RECORD LOAD VS DEFLECTION • REPEAT FOR 24 SPECIMENS TEST 4 — CYCLIC DAMAGE GROWTH TEST AT 70°F • TEST 4 IS THE SAME AS TEST 2 EXCEPT THAT THERE WILL BE DAMAGE IN EACH SPECIMEN PRIOR TO THE START OF TESTING • TEST CONDITIONS AND TEST PROCEDURES ARE THE SAME AS IN TEST 2

Program Element IV, Cyclic Coupon Tests (Continued)

TEST ARTICLE DESCRIPTION	TEST PLAN
TEST SPECIMEN DESCRIPTION (TESTS 1 THROUGH 5)	TEST 5 — CYCLIC DAMAGE GROWTH TEST IN ENVIRONMENT • TEST 5 IS THE SAME AS TEST 3 EXCEPT THAT THERE WILL BE DAMAGE IN EACH SPECIMEN PRIOR TO THE START OF TESTING • TEST CONDITIONS AND TEST PROCEDURES ARE THE SAME AS IN TEST 3

Program Element IV, Cyclic Coupon Tests (Continued)

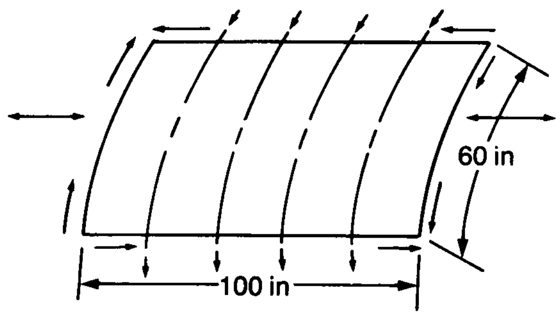
TEST ARTICLE DESCRIPTION	TEST PLAN

TEST SPECIMEN DESCRIPTION (TESTS 6 THROUGH 10)

TYPICAL SKIN PANEL   TYPICAL WINDOW FRAME PANEL
• 2 PANEL CONFIGURATIONS
• 3 PANELS EACH CONFIGURATION FOR EACH TEST
• PANELS ARE FLAT
	TEST PROCEDURE — TESTS 6 THROUGH 10
	• PANELS WILL BE TESTED BY APPLYING SHEAR LOADS IN A STANDARD PICTURE FRAME
	• EACH TEST PANEL WILL HAVE 18 AXIAL STRAIN GAGES, 8 ROSETTE STRAIN GAGES, AND 2 EDIs
	• ALL INSTRUMENTATION CHANNELS WILL BE RECORDED CONTINUOUSLY TO FAILURE DURING THE STATIC TEST
	• EXCEPT AS NOTED, THE TEST PROCEDURE, TEST ENVIRONMENT, NUMBER OF CYCLES, ETC., ARE AS FOLLOWS:
	  • TEST 6 SAME AS TEST 1
	  • TEST 7 SAME AS TEST 2
	  • TEST 8 SAME AS TEST 3
	  • TEST 9 SAME AS TEST 4
	  • TEST 10 SAME AS TEST 5

Program Element IV, Cyclic Panel Tests

TEST ARTICLE DESCRIPTION

TEST 1 SKIN AND STRINGER REPAIR PANEL

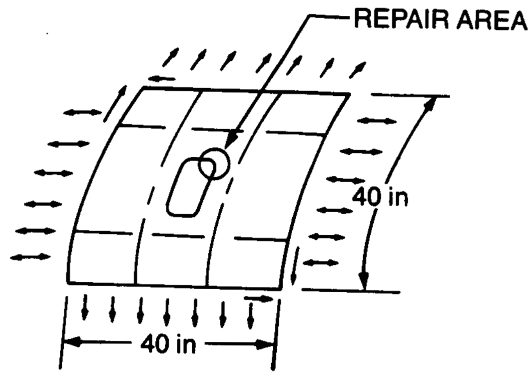


- 12 PANELS — 3 AREA CONFIGURATIONS
- CURVED PANELS — 74-in RADIUS
- PANELS ARE IDENTICAL TO PANELS TESTED IN TEST 11, PROGRAM ELEMENT I, EXCEPT THAT EACH PANEL CONTAINS THREE (3) TYPICAL REPAIRS
- TYPICAL REPAIRS INCLUDED
  - SKIN REPAIR — MECHANICALLY ATTACHED
  - SKIN REPAIR — MECHANICALLY ATTACHED AND BONDED
  - SKIN AND STRINGER REPAIR — MECHANICALLY ATTACHED AND BONDED

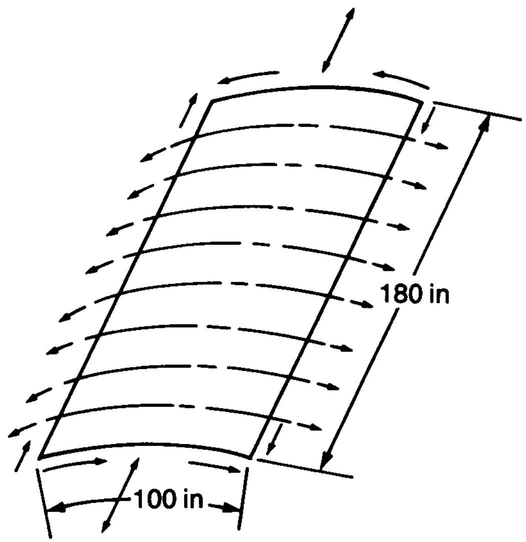
TEST PLAN

- TEST PROCEDURE AND INSTRUMENTATION ARE THE SAME AS FOR TEST 11, PROGRAM ELEMENT I, EXCEPT AS NOTED
- 4 PANELS WILL BE TESTED BY APPLYING COMBINED PRESSURE AND TENSION LOADS TO FAILURE, TWO (2) AT 70°F DRY AND TWO (2) AT −65°F DRY
- 4 PANELS WILL BE TESTED BY APPLYING SHEAR LOADS TO FAILURE, TWO (2) AT 70°F DRY AND TWO (2) AT 160°F WET
- 4 PANELS WILL BE TESTED BY APPLYING COMPRESSION LOADS TO FAILURE, TWO (2) AT 70°F DRY AND TWO (2) AT 160°F WET

Program Element V, Skin and Stringer Repair Panel

TEST ARTICLE DESCRIPTION	TEST PLAN
TEST 2 WINDOW FRAME REPAIR PANEL  • 6 PANELS — 1 CONFIGURATION • CURVED PANEL — 74-in RADIUS • 2 FRAMES — 2 STRINGERS • PANELS ARE IDENTICAL TO PANELS TESTED IN TEST 12, PROGRAM ELEMENT I, EXCEPT THAT ALL PANELS CONTAIN A TYPICAL REPAIR IN THE CRITICAL CORNER OF THE WINDOW FRAME	• TEST PROCEDURE AND INSTRUMENTATION ARE THE SAME AS FOR TEST 12, PROGRAM ELEMENT I, EXCEPT AS NOTED • PANELS WILL BE TESTED UNDER THE CRITICAL COMBINATION OF PRESSURE, SHEAR, AND TENSION OR COMPRESSION LOADS • 2 PANELS WILL BE TESTED AT EACH OF THE FOLLOWING TEMPERATURES: • 70°F DRY • 160°F WET • −65°F DRY

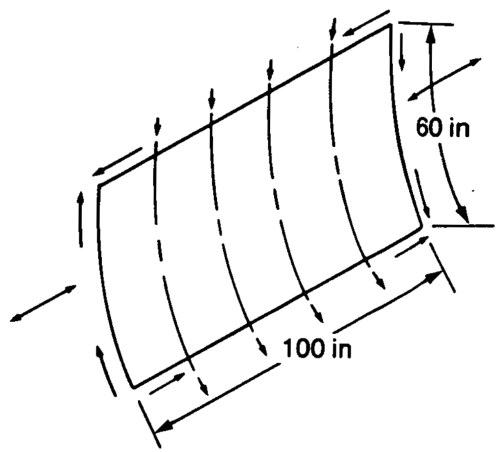
Program Element V, Window Frame Repair Panel

<table>
<tr><th>TEST ARTICLE DESCRIPTION</th><th>TEST PLAN</th></tr>
<tr><td>TEST 1 PRESSURE DAMAGE CONTAINMENT


• 2 PANELS — SAME CONFIGURATION
• 8 FRAMES
• 9 STRINGERS
• 74-in RADIUS</td>
<td>• 30 AXIAL GAGES, 30 ROSETTE GAGES, 10 DEFLECTION GAGES
• PANELS WILL BE TESTED IN COMBINED LOAD TENSION OR COMPRESSION, SHEAR, AND PRESSURE
• EACH PANEL WILL BE LOADED TO A SELECTED LOAD COMBINATION AND A STRAIN SURVEY OBTAINED FOR THE FOLLOWING CONDITIONS:
  • PRESSURE
  • TENSION AXIAL AND SHEAR LOADS
  • TENSION AXIAL AND SHEAR LOADS PLUS PRESSURE
  • COMPRESSION AXIAL AND SHEAR LOADS
  • COMPRESSION AXIAL AND SHEAR LOADS PLUS PRESSURE
• THE STRAIN SURVEYS WILL BE REVIEWED AND COMPARED TO PREDICTED LOAD DISTRIBUTIONS
• SIX DAMAGE TOLERANCE GUILLOTINE TESTS WILL BE CONDUCTED (THREE PER PANEL)
• DAMAGE TOLERANCE GUILLOTINE TESTS WILL BE PERFORMED AS FOLLOWS:
  • LOAD PANEL TO A SPECIFIED COMBINATION OF AXIAL AND SHEAR LOADS AND PRESSURE
  • A 12-in GUILLOTINE BLADE WILL BE SHOT THROUGH THE PANEL AT A FRAME LOCATION
• THE PANEL WILL BE REPAIRED AND A SECOND TEST PERFORMED AT A DIFFERENT LEVEL OF LOADS AND PRESSURE
  • THIS PROCEDURE WILL BE REPEATED AND A THIRD TEST PERFORMED
  • THE FIRST PANEL WILL BE TESTED AT 70°F. THE SECOND PANEL WILL BE TESTED AT −65°F (DRY)</td></tr>
</table>

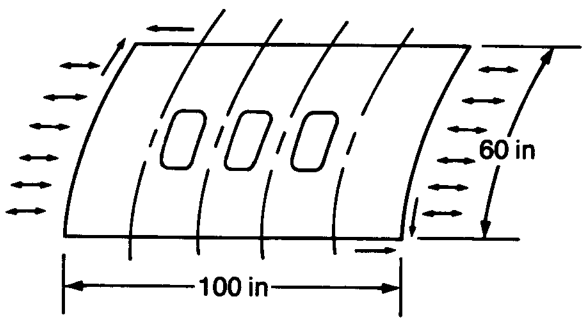
Program Element VI, Pressure Damage Containment

TEST ARTICLE DESCRIPTION	TEST PLAN

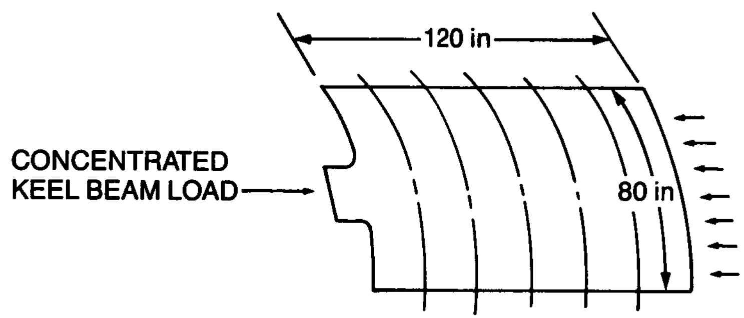
TEST 2 COMPRESSION DAMAGE CONTAINMENT

 	
• 2 PANELS — SAME CONFIGURATION
• 4 FRAMES
• 5 STRINGERS
• 74-in RADIUS
	• 30 AXIAL GAGES, 30 ROSETTE GAGES, 10 DEFLECTION GAGES
	• PANELS WILL BE TESTED IN COMBINED LOAD COMPRESSION AND SHEAR LOADS
	• EACH PANEL WILL BE LOADED TO A SELECTED LOAD LEVEL AND A STRAIN SURVEY OBTAINED FOR THE FOLLOWING CONDITIONS:
	  • COMPRESSION
	  • SHEAR
	  • COMPRESSION AND SHEAR
	• THE STRAIN SURVEYS WILL BE REVIEWED AND COMPARED TO PREDICTED LOAD DISTRIBUTIONS
	• SIX DAMAGE TOLERANCE GUILLOTINE TESTS WILL BE CONDUCTED (THREE PER PANEL)
	• DAMAGE TOLERANCE GUILLOTINE TESTS WILL BE PERFORMED AS FOLLOWS:
	  • LOAD PANEL TO A SPECIFIED LEVEL OF COMPRESSION AND SHEAR LOADS
	  • A 12-in GUILLOTINE BLADE WILL BE SHOT THROUGH THE PANEL AT A STRINGER LOCATION
	• THE PANEL WILL BE REPAIRED AND A SECOND TEST PERFORMED AT A DIFFERENT LEVEL OF COMPRESSION AND SHEAR LOADS
	  • THIS PROCEDURE WILL BE REPEATED AND A THIRD TEST PERFORMED
	  • THE FIRST PANEL WILL BE TESTED AT 70°F. THE SECOND PANEL WILL BE TESTED AT +160°F (WET)

Program Element VI, Compression Damage Containment

TEST ARTICLE DESCRIPTION	TEST PLAN

TEST 3 WINDOW PANEL

• 2 PANELS — SAME CONFIGURATION
• 3 WINDOWS
• 4 FRAMES
• 74-in RADIUS
	• 30 AXIAL GAGES, 30 ROSETTE GAGES, AND 8 DEFLECTION GAGES
	• TEST BOTH PANELS AS FOLLOWS:
	  • CONDUCT FOUR STRAIN SURVEYS AT 70°F
	    • PRESSURE
	    • SHEAR
	    • TENSION
	    • COMPRESSION
	  • REPEAT STRAIN SURVEYS AT − 65°F (DRY)
	  • REPEAT STRAIN SURVEYS AT + 160°F (WET)
	• REVIEW STRAIN SURVEYS
	  • COMPARE TO PREDICTED STRAIN DISTRIBUTION
	  • SELECT COMBINED LOAD FOR ULTIMATE LOAD TEST
	• FIRST PANEL TESTED TO FAILURE AT 70°F WITH SELECTED LOAD COMBINATION
	• SECOND PANEL TESTED TO FAILURE AT + 160°F (WET) WITH SAME LOAD COMBINATION AS THE FIRST PANEL

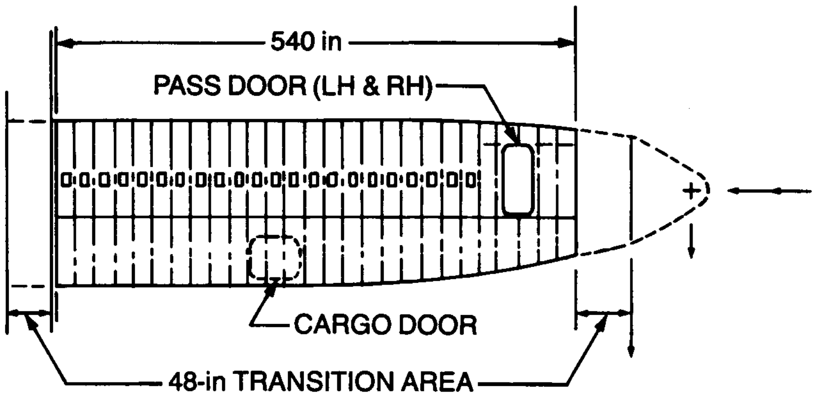
Program Element VI, Window Panel

<table>
<tr><th>TEST ARTICLE DESCRIPTION</th><th>TEST PLAN</th></tr>
<tr><td>
TEST 4 KEEL BEAM LOAD REDISTRIBUTION


• 2 PANELS — SAME CONFIGURATION
• 6 FRAMES
• 9 STRINGERS
• 74-in RADIUS
</td><td>
• 30 AXIAL GAGES, 30 ROSETTE GAGES, 10 DEFLECTION GAGES
• LATERAL SUPPORTS WILL BE PROVIDED AT EACH FRAME
• STRAIN SURVEYS WILL BE CONDUCTED AT 70°F (DRY) AND +160°F (WET) FOR BOTH PANELS
• STATIC TEST FIRST PANEL TO FAILURE AT 70°F
• STATIC TEST SECOND PANEL TO FAILURE AT 160°F (WET)
</td></tr>
</table>

Program Element VI, Keel Beam Load Redistribution

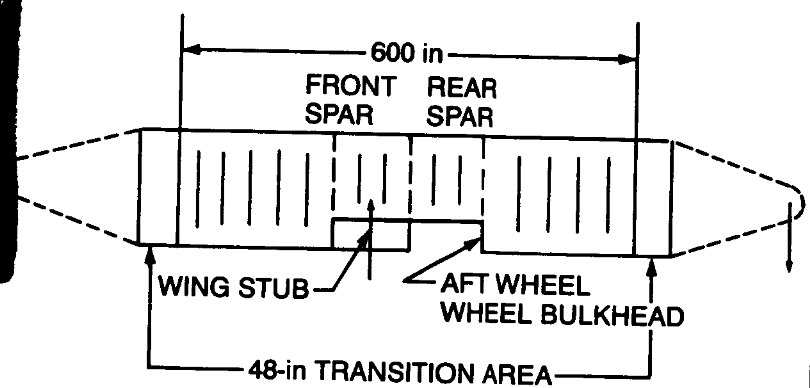
TEST ARTICLE DESCRIPTION	TEST PLAN

FULL-SCALE AFTBODY TEST

• TEST SECTION SIMILAR TO 757 SECTION
• 540-in LONG
• 22 WINDOWS PER SIDE
• 2 PASSENGER DOORS
• 1 CARGO DOOR
• 400 AXIAL STRAIN GAGES, 125 ROSETTE GAGES, 20 DEFLECTION MONITORS
• DISTRIBUTED LOAD APPLIED ALONG LENGTH OF TEST SECTION TO SIMULATE WEIGHT INERTIA LOADING
	LOAD SURVEY
	• 8 LIMIT LOAD TESTS
	
	DURABILITY TEST
	• LOAD LEVELS AND FLIGHT SPECTRUM SIMILAR TO 757 MAJOR FATIGUE TEST
	• TEST AS FOLLOWS:
	  • 8 LOAD SURVEYS
	  • CYCLE FOR 25,000 FLIGHTS (WITHOUT KNOWN DAMAGE)
	  • INFLICT SMALL AREA, NONDETECTABLE DAMAGE AT 8 LOCATIONS
	  CONTINUE CYCLING TO 50,000 FLIGHTS
	  MONITOR AND RECORD ANY DAMAGE GROWTH
	  • INFLICT SMALL AREA, DETECTABLE DAMAGE AT 6 LOCATIONS
	  CONTINUE CYCLING TO 100,000 FLIGHTS
	  MONITOR AND RECORD ANY DAMAGE GROWTH
	
	STATIC TEST
	• 4 DAMAGE TOLERANCE (GUILLOTINE) TESTS
	• 3 ULTIMATE LOAD TESTS
	• STATIC TEST TO FAILURE

Program Element VII, Full-Scale Aftbody

TEST ARTICLE DESCRIPTION	TEST PLAN
FULL-SCALE CENTER SECTION	LOAD SURVEY

FULL-SCALE CENTER SECTION



- TEST SECTION SIMILAR TO 757 SECTION
- 600-in LONG
- 30 WINDOWS PER SIDE
- 2 OVERWING EXIT DOORS
- 2 CARGO DOORS
- 400 AXIAL STRAIN GAGES, 125 ROSETTE GAGES, 20 DEFLECTION MONITORS
- DISTRIBUTED LOAD APPLIED ALONG LENGTH OF TEST SECTION TO SIMULATE WEIGHT INERTIA LOADING

TEST PLAN

LOAD SURVEY

- 8 LIMIT LOAD TESTS

DURABILITY TEST

- LOAD LEVELS AND FLIGHT SPECTRUM SIMILAR TO 757 MAJOR FATIGUE TEST
- TEST AS FOLLOWS:
  - 8 LOAD SURVEYS
  - CYCLE FOR 25,000 FLIGHTS (WITHOUT KNOWN DAMAGE)
  - INFLICT SMALL AREA, NONDETECTABLE DAMAGE AT 8 LOCATIONS
    CONTINUE CYCLING TO 50,000 FLIGHTS
    MONITOR AND RECORD ANY DAMAGE GROWTH
  - INFLICT SMALL AREA, DETECTABLE DAMAGE AT 6 LOCATIONS
    CONTINUE CYCLING TO 100,000 FLIGHTS
    MONITOR AND RECORD ANY DAMAGE GROWTH

STATIC TEST

- 4 DAMAGE TOLERANCE (GUILLOTINE) TESTS
- 3 ULTIMATE LOAD TESTS
- STATIC TEST TO FAILURE

Program Element VIII, Full-Scale Center Section

1. Report No. NASA CR 172406	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle STUDY ON UTILIZATION OF ADVANCED COMPOSITES IN FUSELAGE STRUCTURES OF LARGE TRANSPORTS		5. Report Date FEBRUARY 1985
		6. Performing Organization Code B8600
7. Author(s) JOHNSON, R. W., THOMSON, L. W., WILSON, R. D.		8. Performing Organization Report No.
		10. Work Unit No.
9. Performing Organization Name and Address BOEING COMMERCIAL AIRPLANE COMPANY P.O. BOX 3707 SEATTLE, WASHINGTON 98124		11. Contract or Grant No. NAS1-17417
		13. Type of Report and Period Covered CONTRACTOR REPORT
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS AND SPACE ADMINISTRATION WASHINGTON, D. C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

LANGLEY TECHNICAL MONITOR: JON S. PYLE
FINAL REPORT

16. Abstract

A study was performed to assess the potential for utilizing advanced composites in fuselage structures of large transports. Six fuselage design concepts were selected and evaluated in terms of structural performance, weight, and manufacturing development and costs. Two concepts were selected that merit further consideration for composite fuselage application. These concepts are (1) a full depth honeycomb design with no stringers, and (2) an I-section stringer stiffened laminate skin design. Weight reductions due to applying composites to the fuselages of commercial and military transports were calculated. The benefits of applying composites to a fleet of military transports were determined. Significant technology issues pertinent to composite fuselage structures were identified and evaluated. Program plans for resolving the technology issues were developed. Boeing's preferred option for demonstrating technology readiness was selected.

17. Key Words (Suggested by Author(s)) COMPOSITE STRUCTURE, GRAPHITE-EPOXY, FUSELAGE, WEIGHT REDUCTION, TECHNOLOGY ISSUES, COMMERCIAL TRANSPORT, MILITARY TRANSPORT	18. Distribution Statement [redacted]

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
UNCLASSIFIED	UNCLASSIFIED	153	

N-305 [redacted]